# *Study of* EXPANDABLE,
# TERMINAL DECELERATORS FOR
# MARS ATMOSPHERE ENTRY

*By*

**GOODYEAR AEROSPACE CORPORATION**
AKRON, OHIO

*For*

**JET PROPULSION LABORATORY**
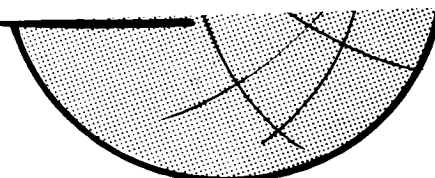JPL CONTRACT NO. 951153

3 October 1966
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# GOODYEAR AEROSPACE

## CORPORATION

AKRON 15, OHIO

STUDY OF EXPANDABLE, TERMINAL

DECELERATORS FOR MARS ATMOSPHERIC ENTRY

Volume II - Supporting Data and Technical Analysis

GER-12842                                    3 October 1966

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for

Jet Propulsion Laboratory, California Institute of Technology

Pasadena, Calif.

# ABSTRACT

The suitable application of expandable terminal decelerators
for Mars atmosphere entry has been established on the basis
of straightforward engineering techniques of analysis and de-
sign. By relating decelerator and entry vehicle performance
and configuration characteristics to constraints associated
with the operating environments, parametric formats were
formulated that provided the required data. From these data
integrated and meaningful tradeoffs were made. Analytical
tools employed were point-mass trajectory computations,
generalized strength/weight and configuration analyses, drag
performances estimates, pressure distribution estimates,
thermal analyses, and aerodynamic stability analyses. The
results of this study have established that the suitable appli-
cation of expandable terminal decelerators for Mars atmo-
sphere entry are within the state of the art of expandable
decelerator technology.

# FOREWORD

The research described in this supporting technical data
report (Volume II) was performed by Goodyear Aerospace
Corporation, subsidiary of The Goodyear Tire & Rubber
Company, Akron, Ohio, for the Jet Propulsion Laboratory,
California Institute of Technology, under the authority of
Contract No. 951153. The work was conducted from December 1965 to October 1966. Mr. James M. Brayshaw, Jr.,
was the Jet Propulsion Laboratory Technical Representative.

The work was performed under the general direction of
Mr. R L. Ravenscraft, manager of the Aero-Mechanical
Engineering Division and Mr. Fred R. Nebiker, manager of
the Recovery Systems Engineering Department. The program was directed by Mr. Jay L. Musil, project engineer.

Personnel contributing to this effort were Mssrs. A. P. Ahart,
configuration, weight, and strength analyses; K. Birklein
and J. W. Schlemmer, computer analyses; I. M. Jaremenko,
pressure distribution analyses; and W. W. Sowa, thermal
analyses.

This is Volume II of two volumes. Volume I presents the
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1. AERODYNAMIC COEFFICIENT DATA

   Figures 1 through 10 include various aerodynamic coefficient data appropriate to the basic entry capsule configuration. These data include results of experimental and theoretical analytical studies conducted by JPL and NASA.

2. BLUNT AERODYNAMIC SHAPES

   To support aerodynamic stability analyses for the Mars terminal decelerator program, Goodyear Aerospace investigated published data applicable to the decelerator configurations under study. Unfortunately, the meager data were of questionable accuracy. No available data relate to effects of flexibility on decelerator stability coefficients at angles of attack or for trailing decelerators. The following graphic information summarizes available data that more nearly relate to the configurations considered in the study.

   Figures 11 through 16 are graphic summaries of published data on blunt aerodynamic shapes.

3. CONES

   Plots of $C_{N_\alpha}$ and cp versus Mach number present a fairly complete picture of the static stability of a cone in free stream. Experimental data given up to Mach 6.8 for cones of large angle $(\theta_s)$, agree with Newtonian theory. Several other theories predict supersonic stability characteristics; these, however, are of limited accuracy for large $\theta_s$.

   The effect of nose bluntness is negligible up to r/D = 0.4 where r = nose radius and D = cone base diameter, especially at $\theta_s \gtreqless 30$ deg.
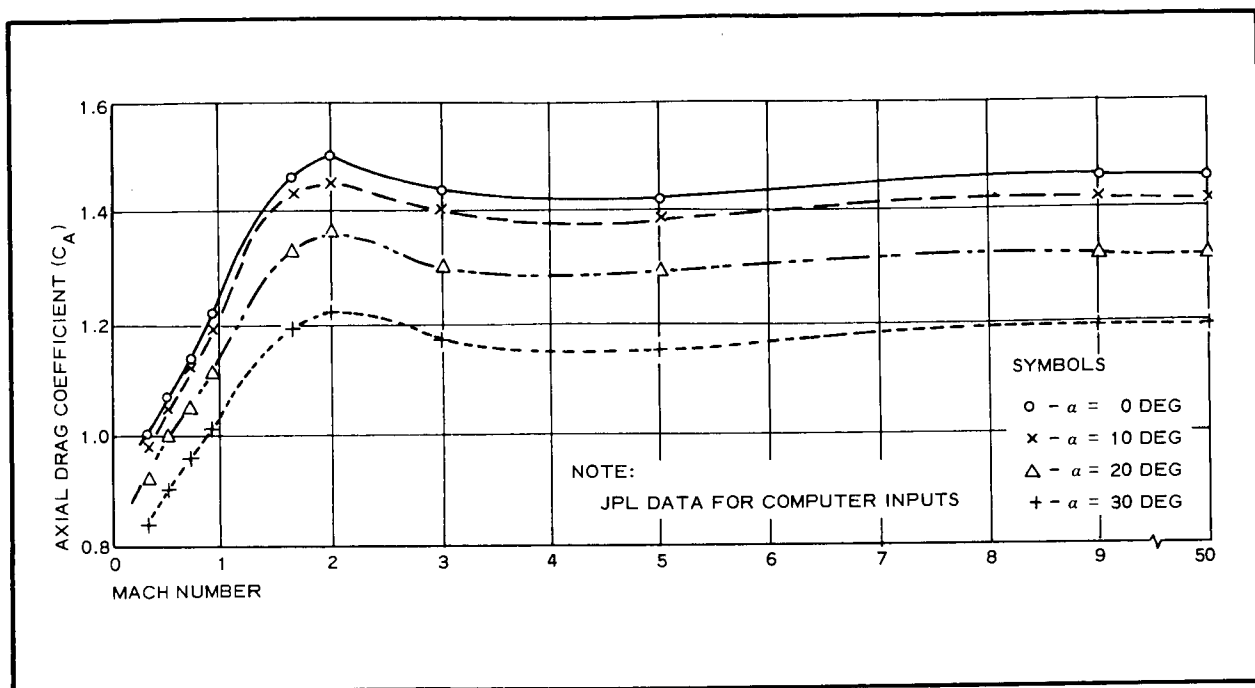
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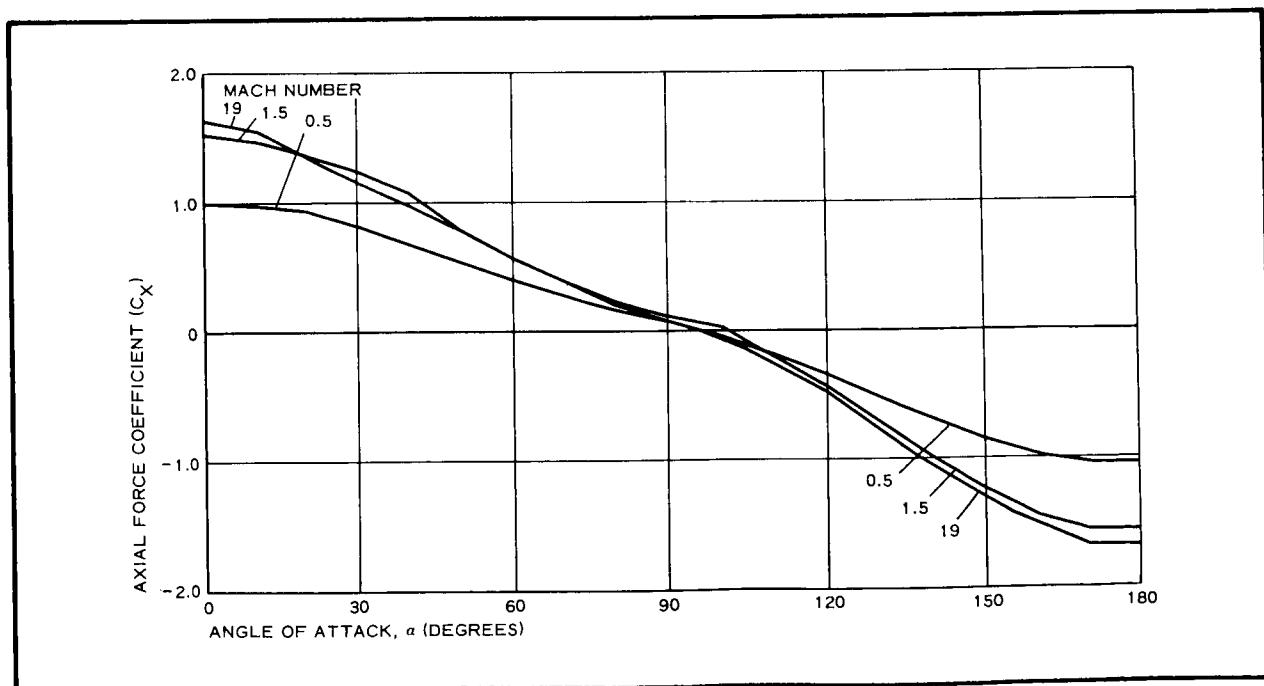


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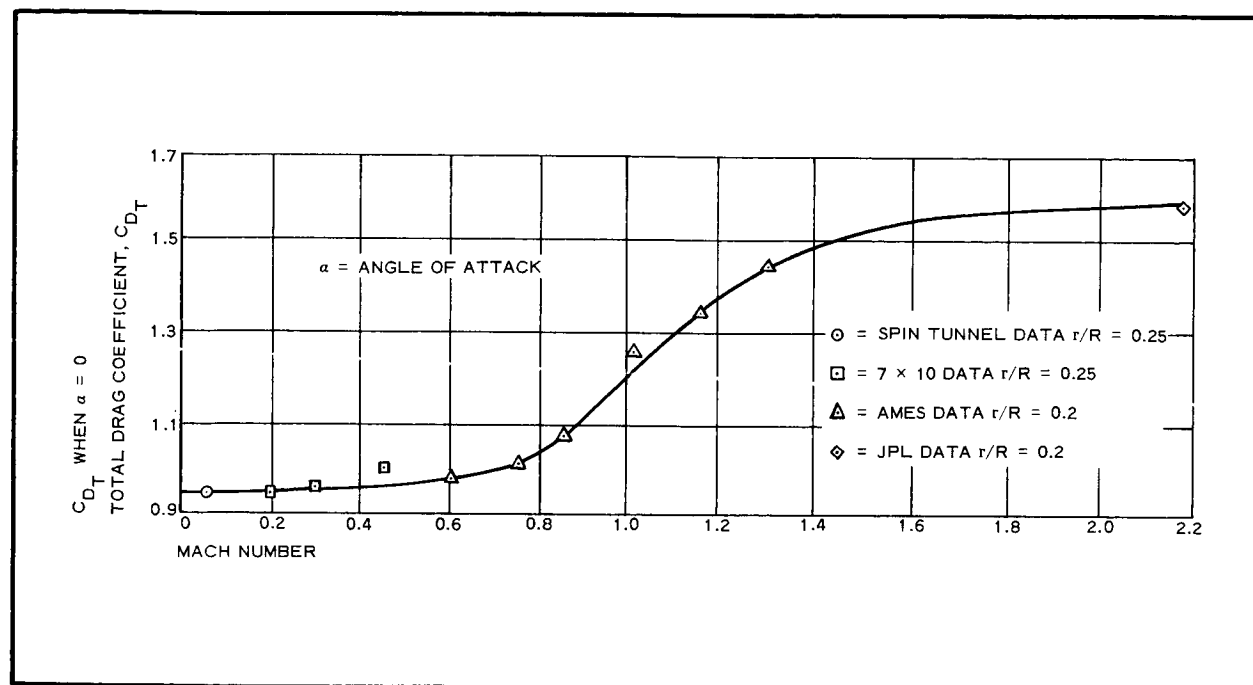
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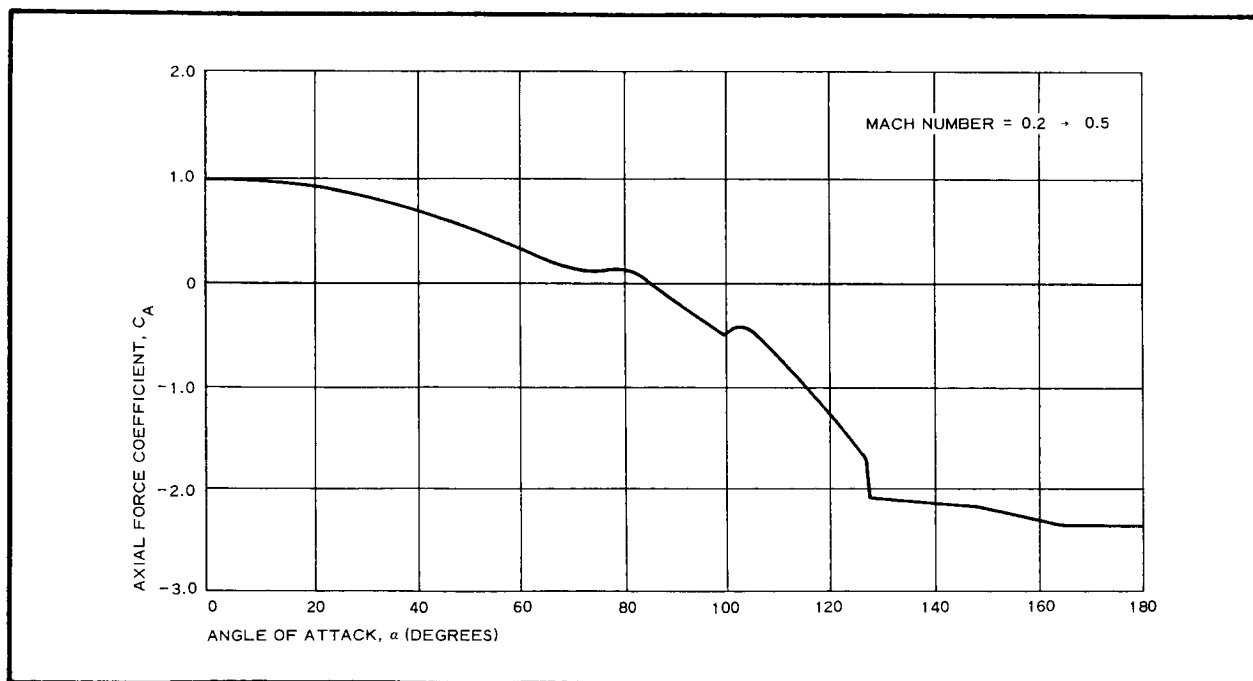


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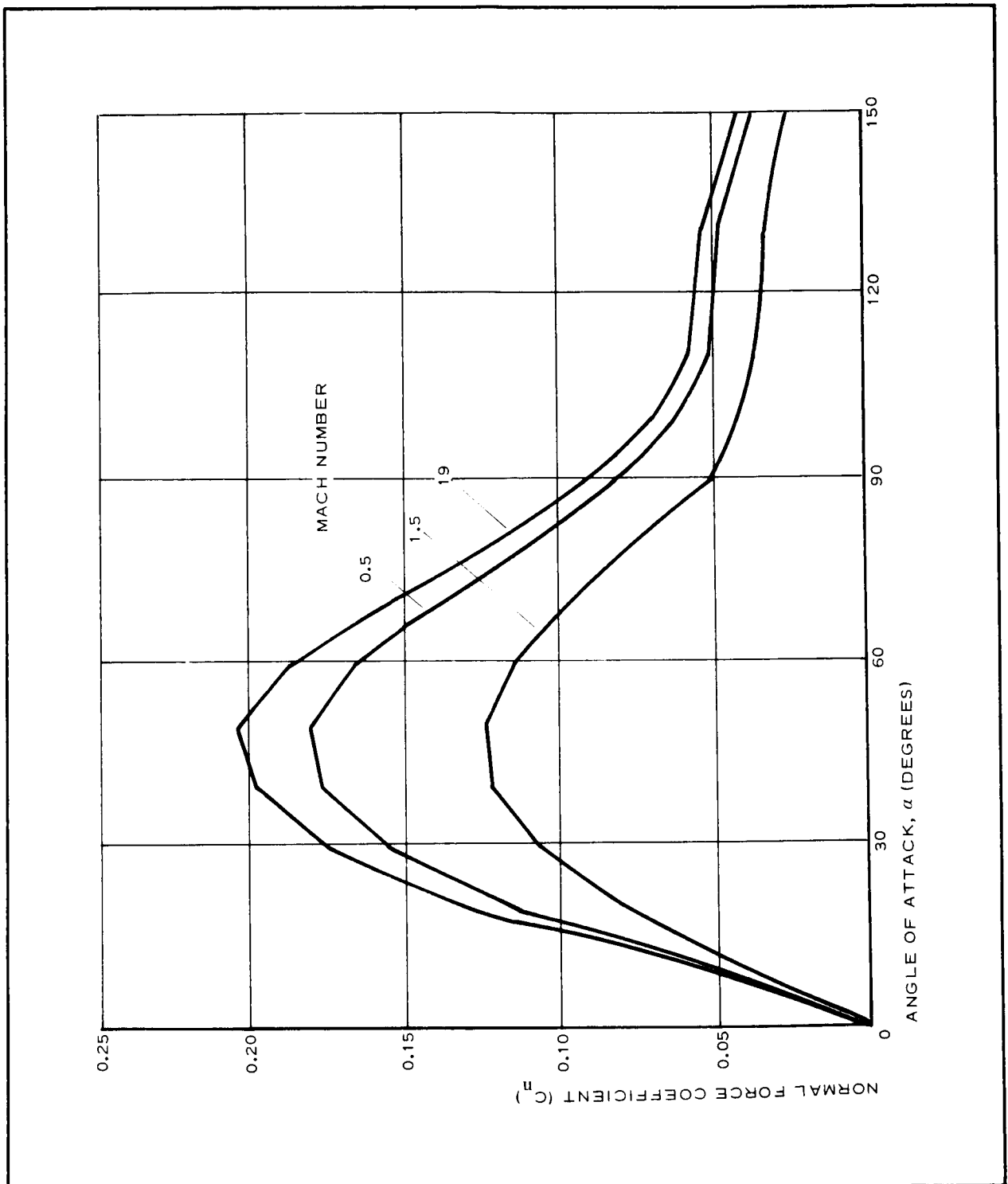
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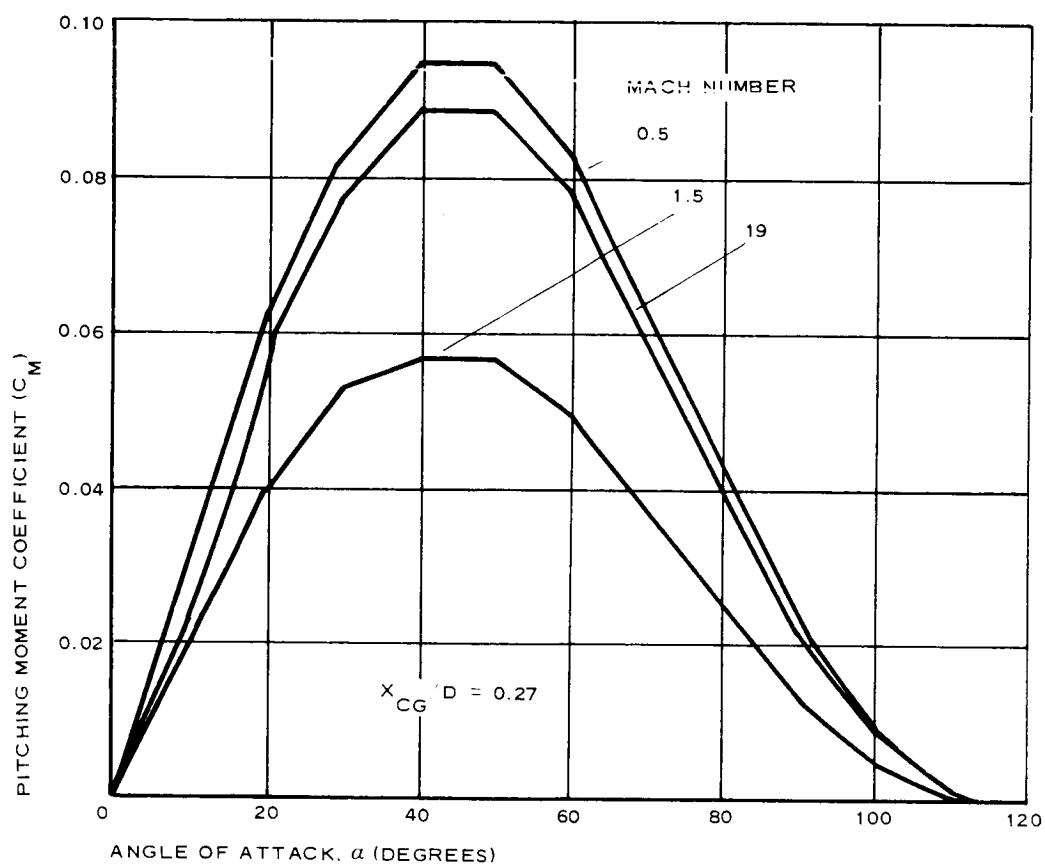
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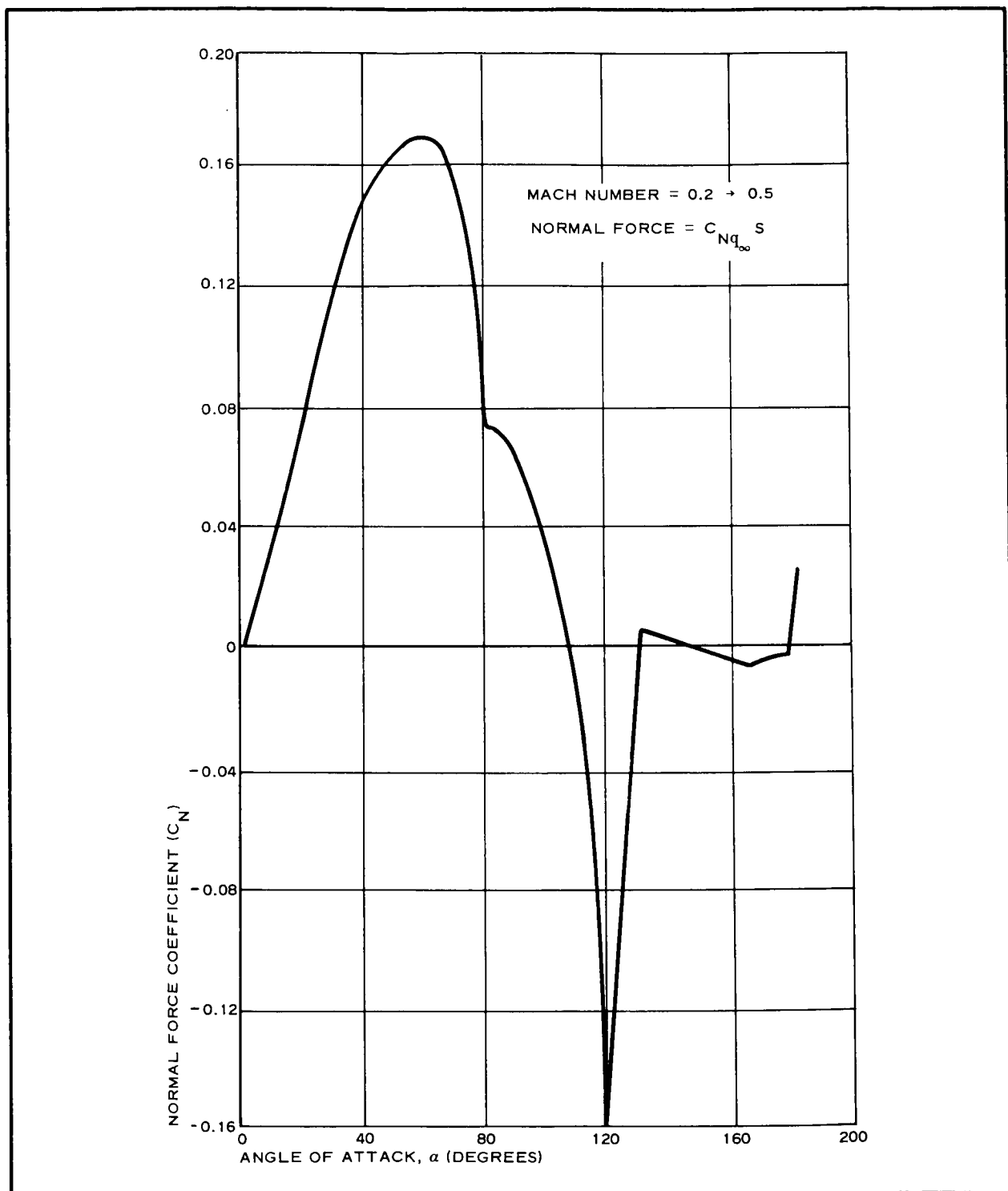
Figure 7 - $C_N$ versus $\alpha$ for 120-Deg Blunt Cone

MACH NO. = 0.2 → 0.5

MOMENT = $C_m q_\infty SD$

MOMENT CENTER:

$\dfrac{X}{D}$ = 0.199 FROM VIRTUAL TIP

$\alpha$ = ANGLE OF ATTACK

MOMENT COEFFICIENT ($C_m$)

$\alpha$ (DEGREES)
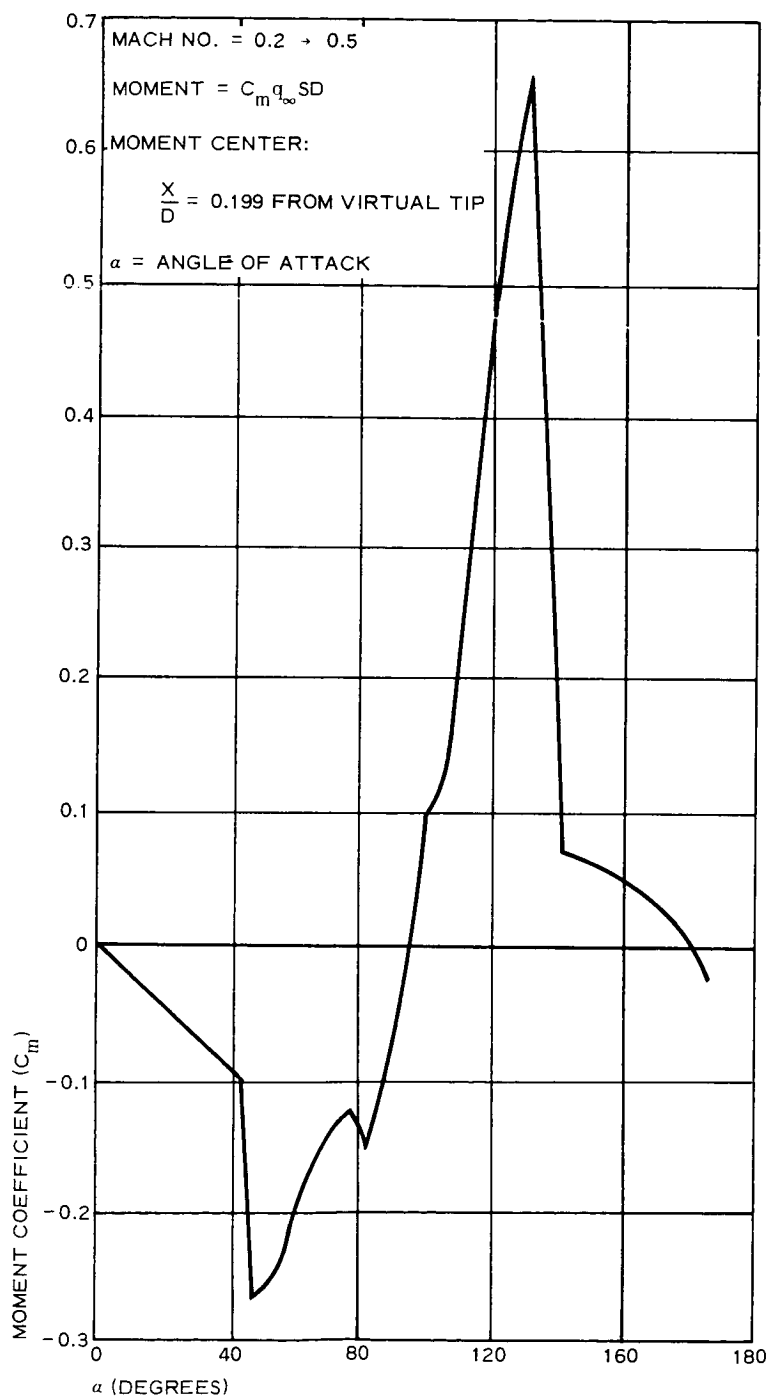
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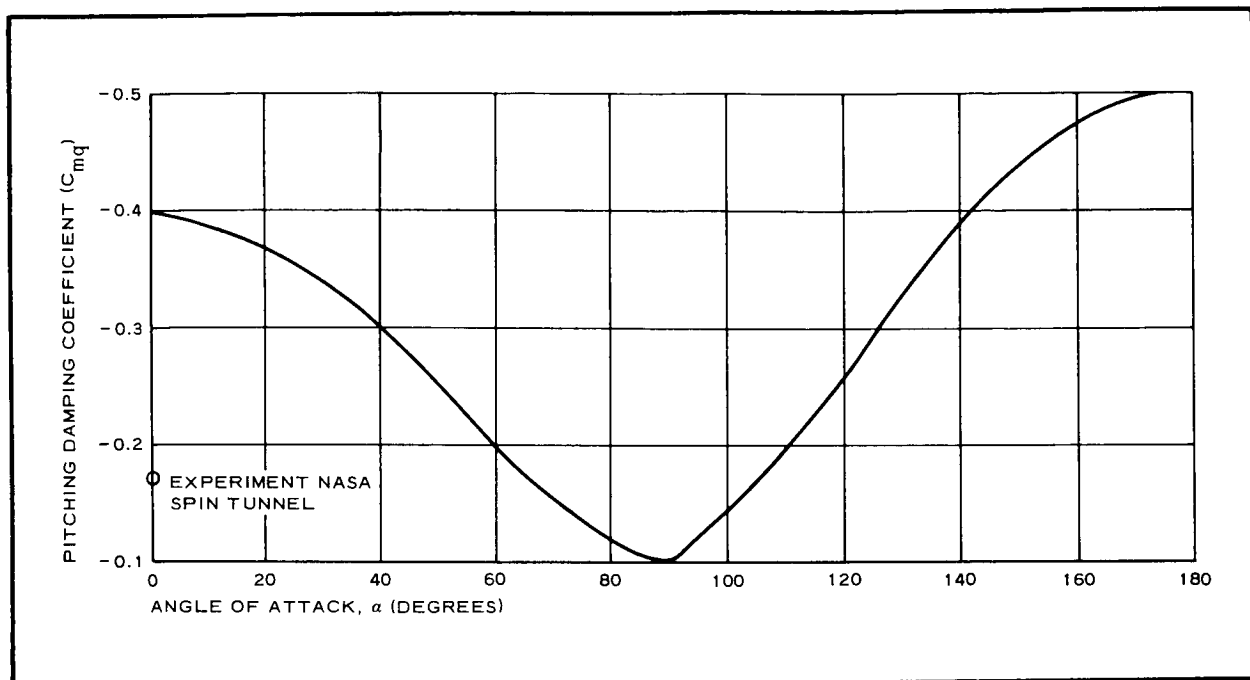
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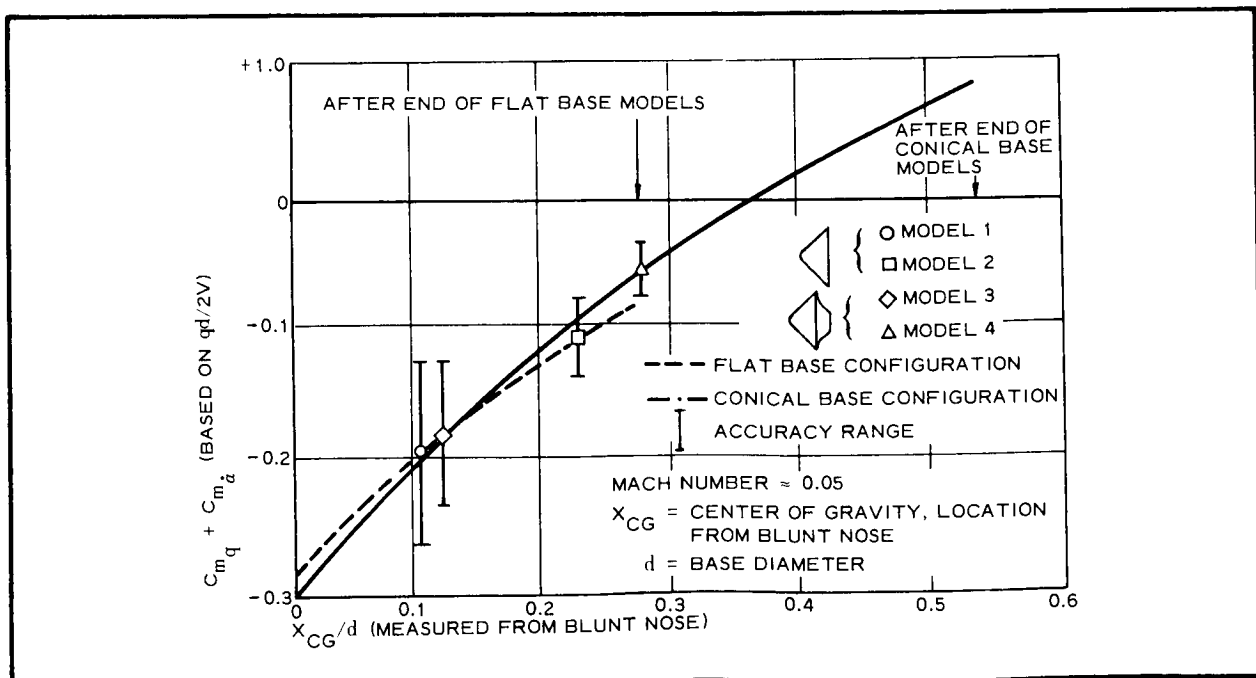


Figure 10 - Variations of $C_{m_q} + C_{m_\alpha}$ for Two Blunt Cone Models

REFERENCE: WADC TR-59-324, PART II

NOTES:

ALL COEFFICIENTS ARE REFERENCED TO S AND D WHEN

$$S = \frac{\pi D^2}{4} \quad \text{AND}$$

$\theta_S$ = FLARE HALF ANGLE
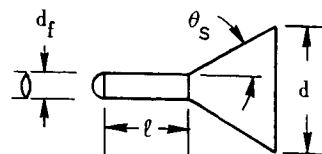
"APPLIES TO FIGURES 11, 12, AND 13."

$\frac{\ell}{d_f}$ = 0.5 FOR A, B AND C

$\frac{\ell}{d_f}$ = 1.72 FOR D, E AND F

* MOMENT CENTER NOT GIVEN IN REFERENCE
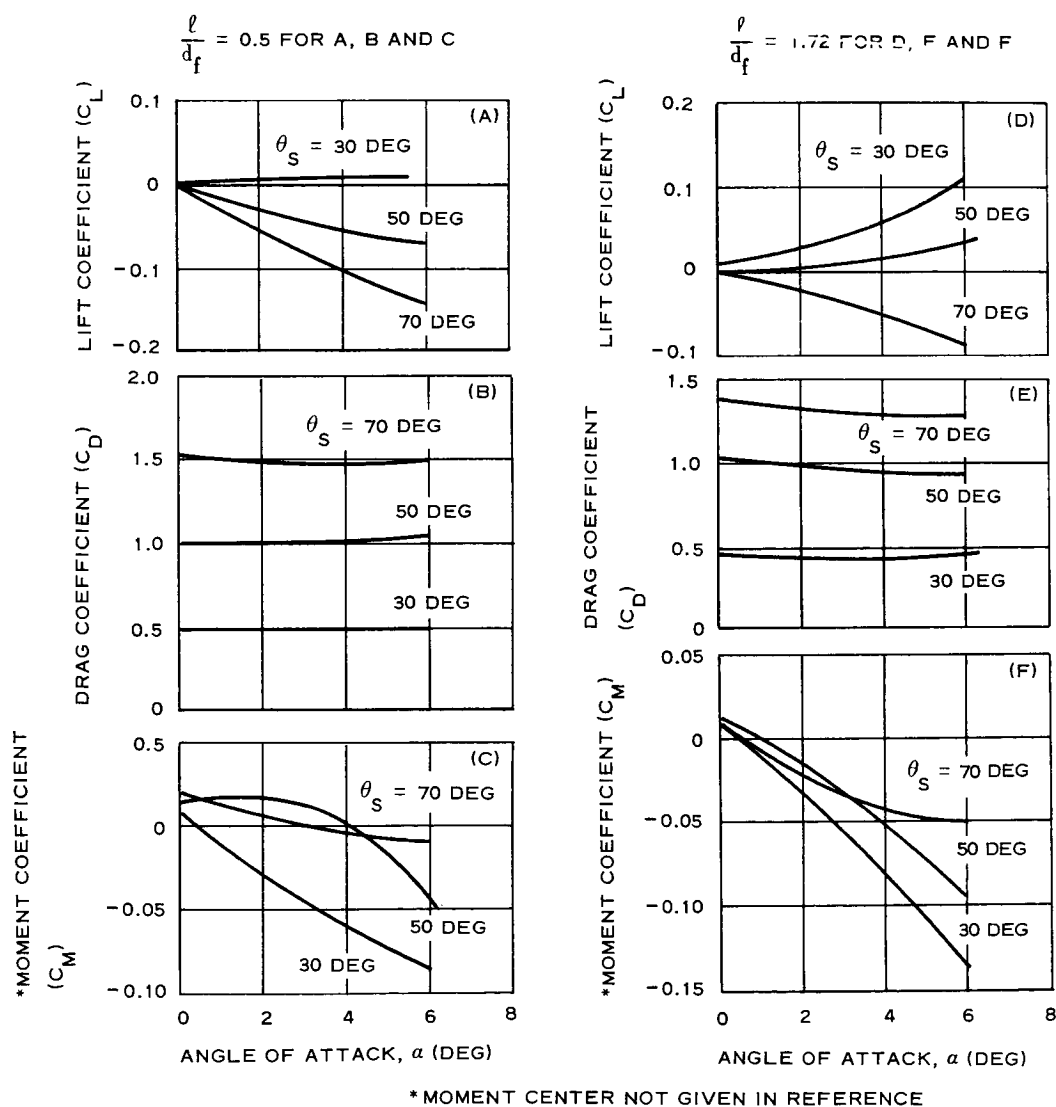
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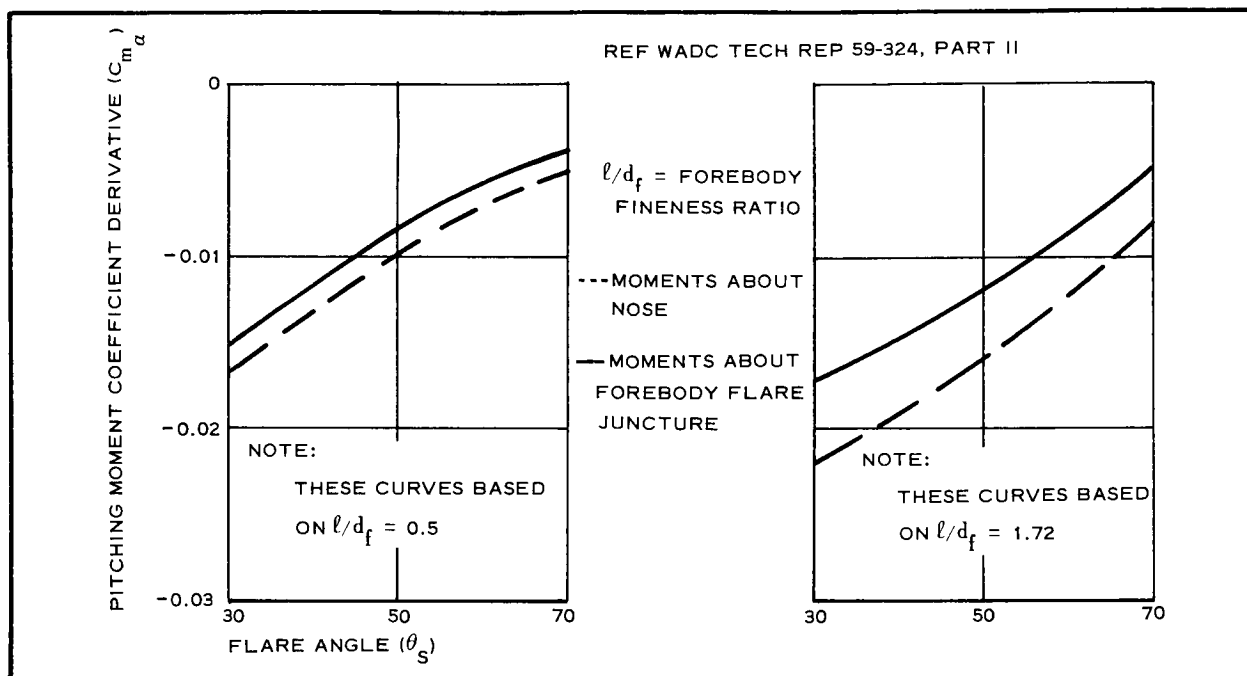
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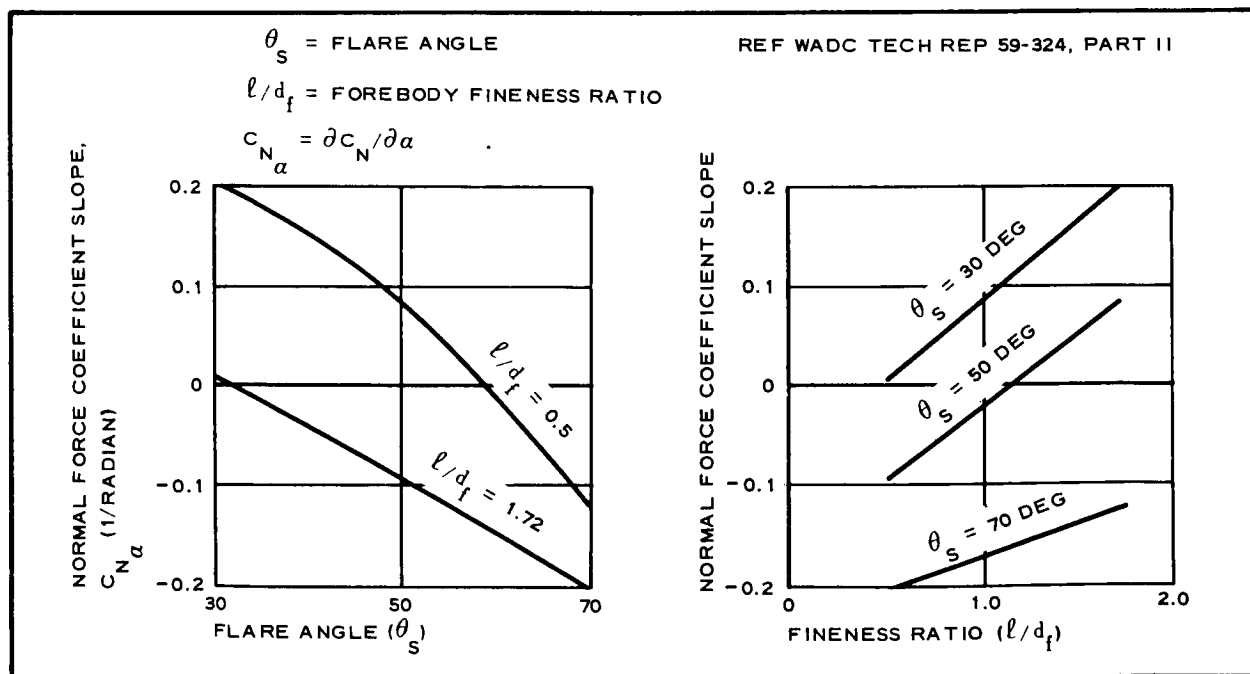


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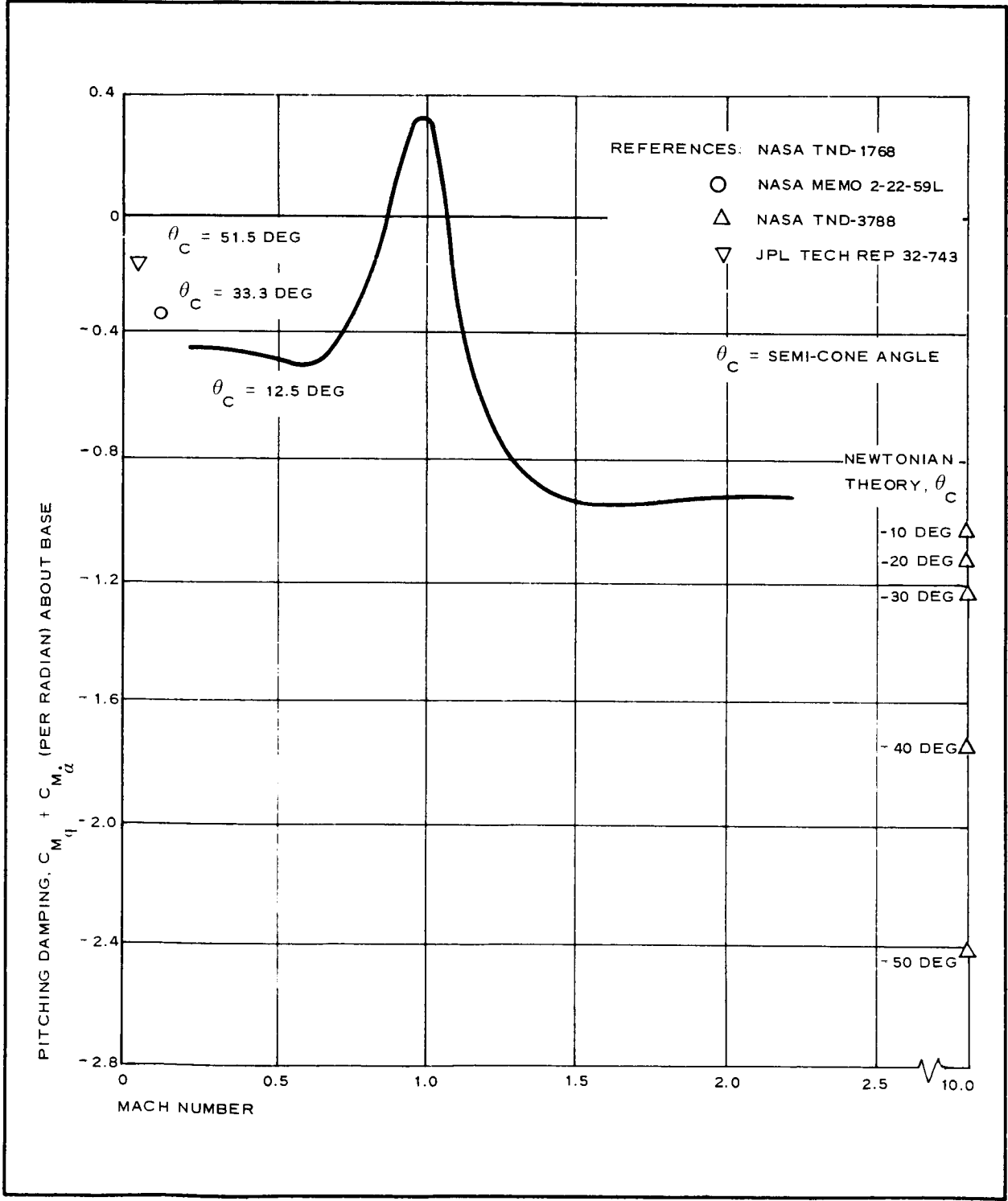
Figure 14 - Damping Moment Characteristics for Cones

Apparently, few, if any, tests have been made to evaluate $R_N$ effects at low speeds.

Dynamic stability is an area where few tests have been conducted with cones of large $\theta_s$. The plot of $C_{m_q} + C_{m_\alpha}$ test data for a cone of $\theta_s =$ 12.5 deg is of questionable accuracy because of error in transferring the moment center to the cone base. The average variation of $C_{m_q} + C_{m_\alpha}$ with Mach number is based on test data about three reference centers. Prior to averaging, the transferred moments did not agree.

Several theories for damping moments at speeds above $M = 1.0$ exist and are in varying agreement with limited test data.

4. FLARED SKIRTS

The only static stability test data applicable to large decelerator-type flares are for Mach = 8.0. Graphs are presented in Figures 11, 12, and 13 for the static stability coefficient variation with flare angle and fore-body fineness ratio.

Much test data have been compiled by various government agencies for small flared skirts used for missile and re-entry body stability at very high speeds. These data, difficult to correlate because of many variations in forebody geometry, are of limited value in decelerator study.

No dynamic stability test data were found for large flare angles and diameters at any speed.

5. TENSION SHELLS

The only stability test data for a tension-shell body came from a free-spinning wind tunnel test at Mach 0.044 and $R_N = 0.4 \times 10^4$ per diameter (see Figure 17).

The following stability data apply to the tension shell shown in Figure 17:

13 ⓐ

$\theta_C$ = SEMIAPEX ANGLE

$$C_{N_\alpha} = \frac{\partial C_N}{\partial a}$$

$$C_N = \frac{N}{\frac{1}{2}\rho V^2 S}$$

$$S = \frac{\pi D^2}{4}$$

$\theta_C$ =

2.0          2.5          3.0
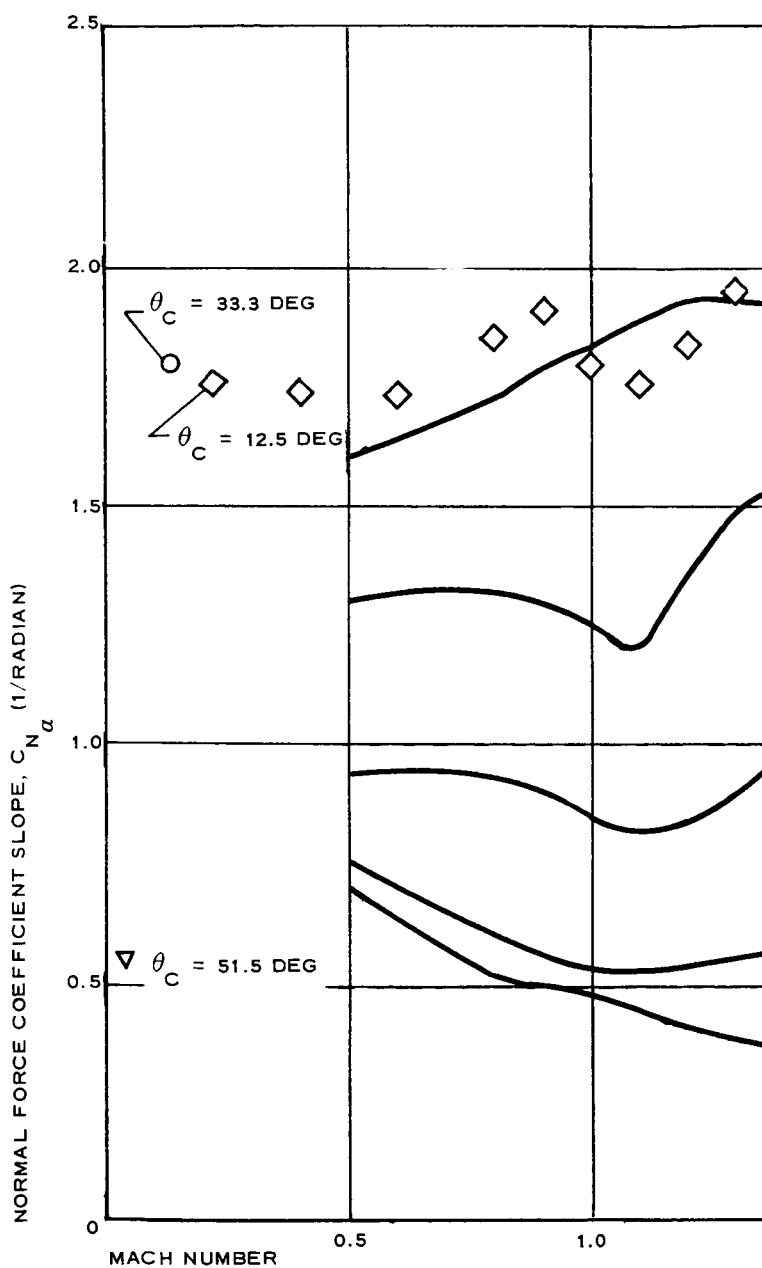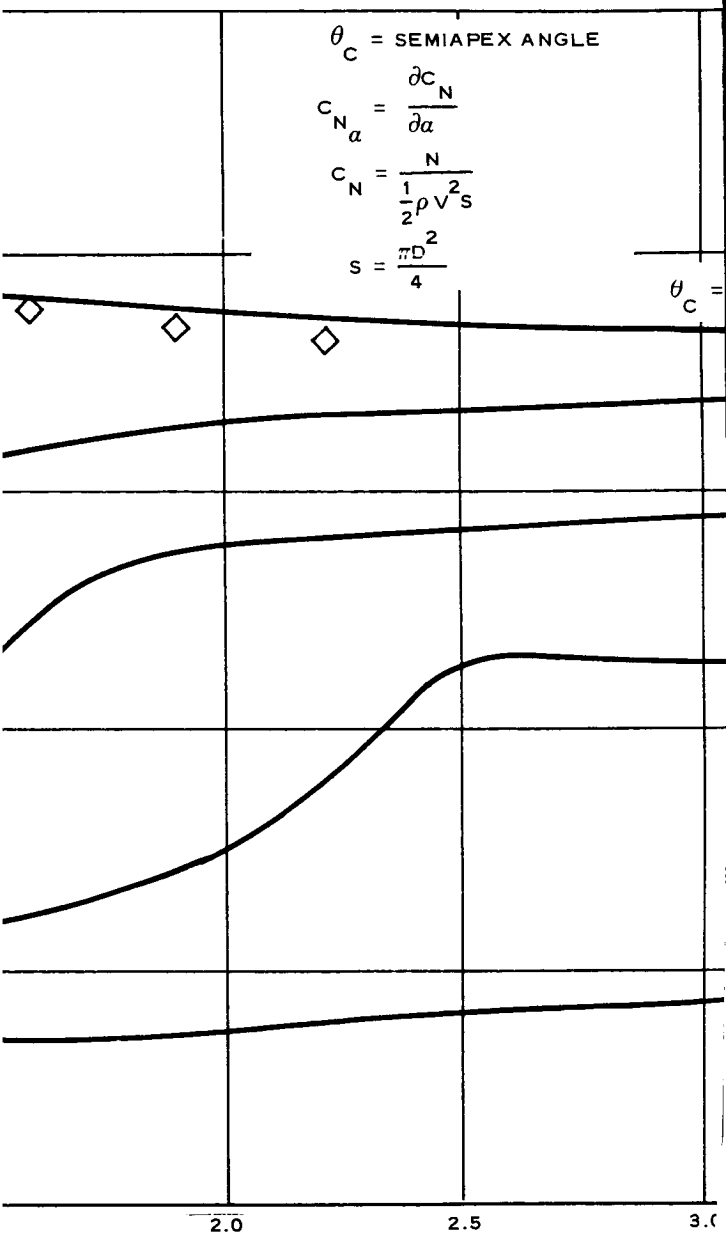
13

②
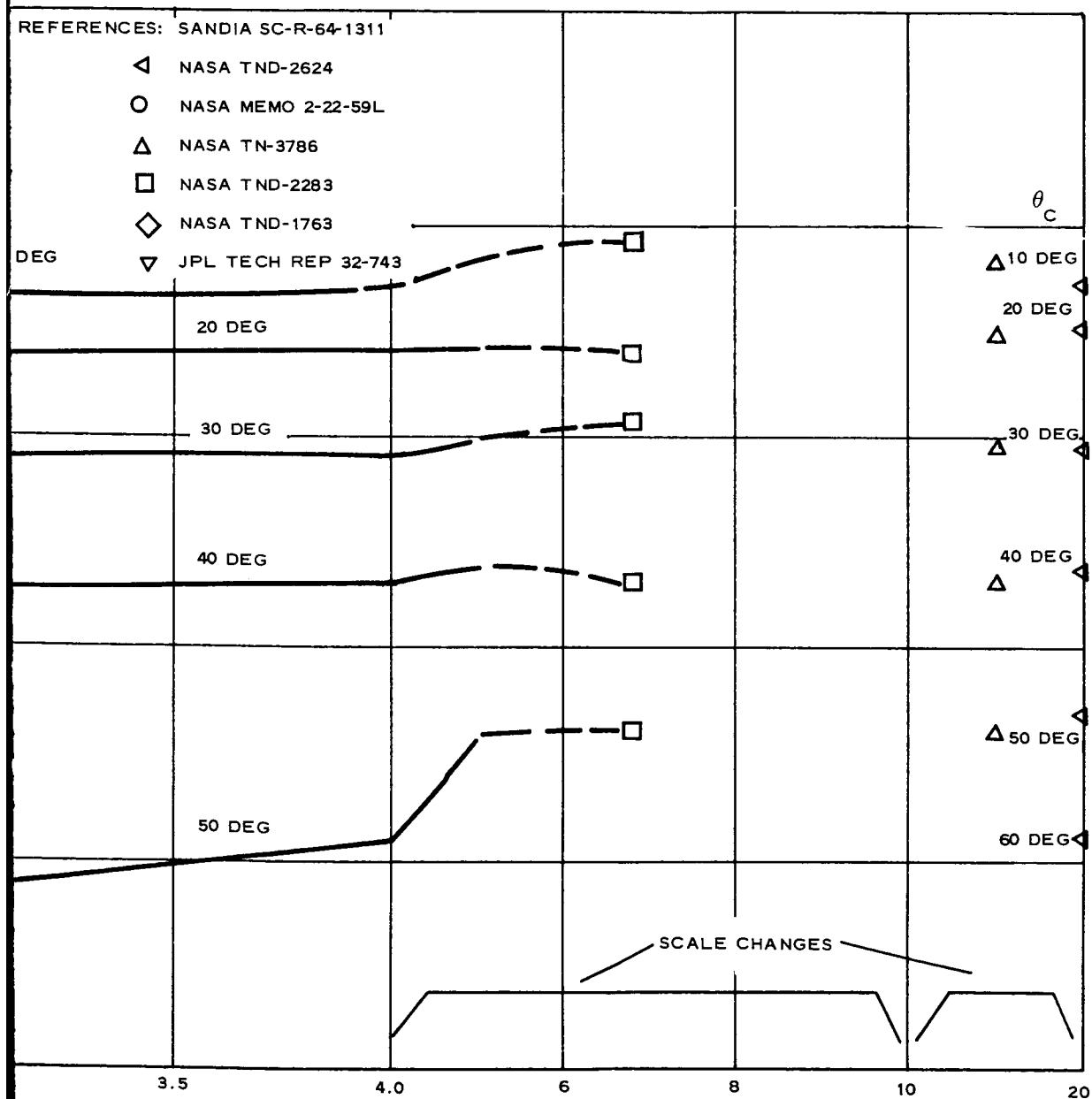
Figure 15 - Static Stability of Cones, Normal Force Curve Slope

Figure showing CENTER OF PRESSURE (x/ℓ) versus MACH NUMBER. The vertical axis ranges from 0.6 to 2.4, and the horizontal axis (MACH NUMBER) ranges from 0 to 3.0. Legend markers: ⊙ $\theta_C = 33.3$ DEG and ◇ 12.5 DEG.

15
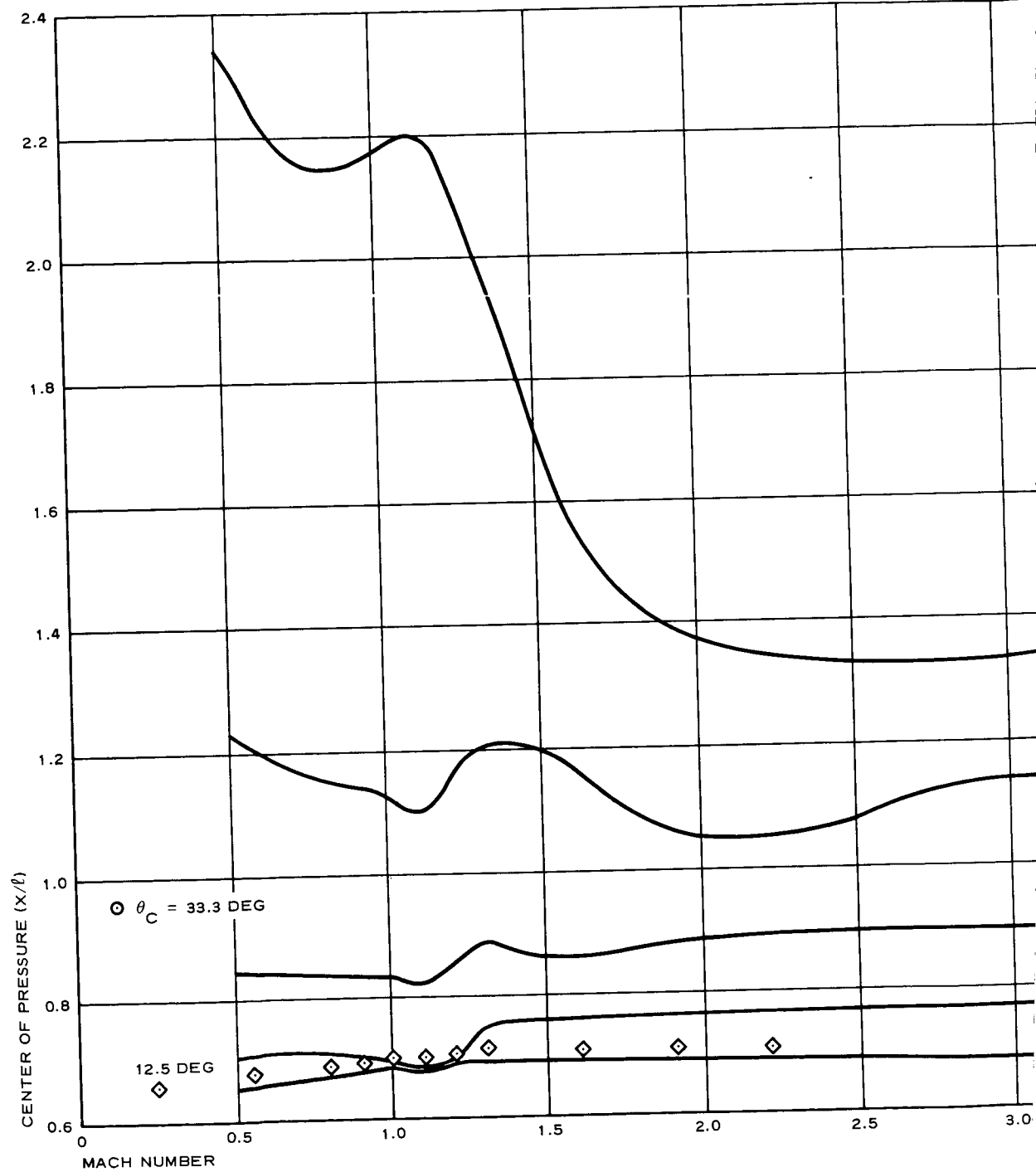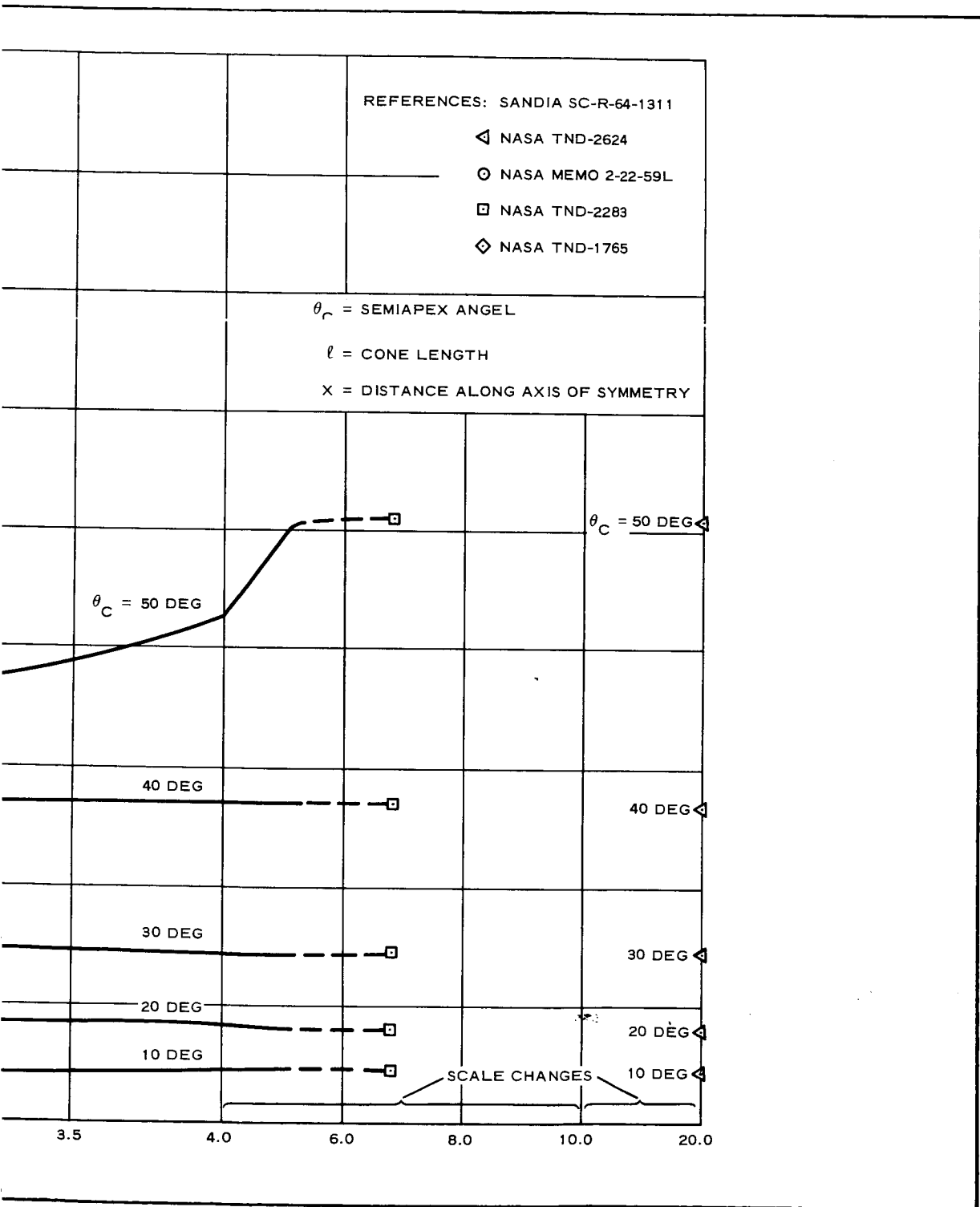
Figure 16 - Static Stability of Cones, Center of Pressure

$$\frac{\ell}{D} = 2.0$$

$$\frac{r}{D} = 0.140$$

$$\frac{d}{D} = 0.280$$

$$\frac{CG}{D} = 0.245$$

$$(\frac{CG}{D} = 0.224\text{-SMALL CG POSITION IN REFERENCE})$$
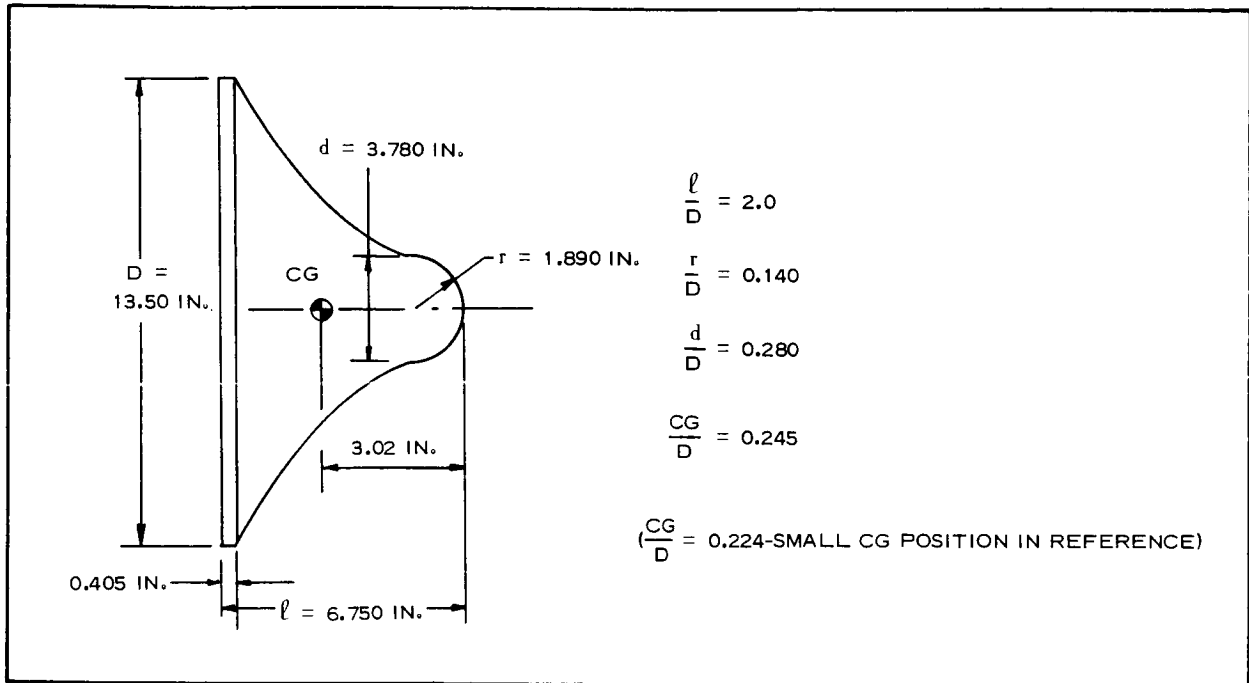
Figure 17 - Tension Shell

$C_{L_\alpha} = -0.577$ per radian

$C_{m_\alpha} = -0.086$ per radian $\Bigg\}$ linear up to $\alpha = 40$ deg

$C_{N_\alpha} = +0.515$ per radian

$C_D = 1.09$

$C_{m_q} = -0.097$ per radian
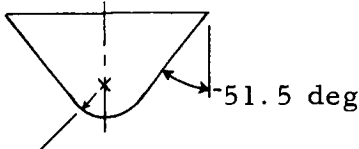
Moment center = cg

$\dfrac{cp}{D} = 0.411$

$\dfrac{cp}{\ell} = 0.822$ $\Bigg\}$ based on cg at 0.245

Coefficients are based on D, $\pi D^2/4$, and $qD/v$.

Table I compares tension-shell aerodynamics with more familiar shapes.

### TABLE I - TENSION-SHELL AERODYNAMICS

| Body shape | $C_{L_\alpha}$ | $C_{m_\alpha}$ | $C_{m_q}$ | $\frac{cp}{\ell}$ from nose | $C_D$ |
|---|---|---|---|---|---|
| Blunted cone <br> ($-51.5$ deg) <br> $\frac{r}{D} = 0.235$ | -0.330 | -0.183 | -0.148 | 0.591 | 0.880 |
| Hemisphere | +0.620 | -0.721 | -0.281 | 0.420 | 0.450 |
| Tension shell | -0.577 | -0.086 | -0.097 | 0.490 | 1.09 |

## SECTION II - AERODYNAMIC AND THERMAL ENVIRONMENT ANALYSES

1.  GENERAL

    The following describes the range of aerodynamic loading and thermal environments the decelerator devices must be designed to face to perform successfully with structural integrity.

2.  ATMOSPHERIC ENVIRONMENT

    The variation of Mars atmospheric density with altitude below the tropopause is assumed to be in accordance with the adiabatic lapse-rate relation:

    $$\rho_h = \rho_o \left(1 - \frac{\Gamma h}{T_o}\right)^{\frac{1}{\gamma - 1}} , \qquad (1)$$

    where

    $\rho_h$ = density at altitude

    $\rho_o$ = surface density,

    $\Gamma$ = adiabatic lapse rate $K^+$/km and R/1000 ft ,

    $h$ = altitude,

    $T_o$ = surface temperature, and

    $\gamma$ = specific heat ratio.

    Variation of static pressure with altitude is assumed to be in accordance with the reversible adiabatic process relation:

$$\frac{P_o}{P_1} = \left(\frac{T_o}{T_1}\right)^{\frac{\gamma}{\gamma - 1}} , \tag{2}$$

where

$P_o$ = surface pressure,

$P_1$ = pressure altitude,

$T_o$ = surface temperature, and

$T_1$ = stratospheric temperature.

Principles of gas dynamics for the inviscid ideal gas are assumed to be valid for the atmospheric properties listed in Table II. From JPL composite initial trajectories for VM7 and VM8 atmospheres, the deployment free-stream Mach numbers and altitudes are shown in Figure 18. For altitudes considered for the decelerator deployment, the variations in ambient pressure, temperature, and speed of sound are evaluated and plotted as bands of magnitudes versus altitude in Figures 19 and 20.

3. SURFACE PRESSURES ON DECELERATORS IN HIGH-SPEED FLOWS

Rapid techniques for estimating pressure distributions on decelerators in the high-speed flows are employed for analyses described in this section. One consists of the Newtonian impact law modified by replacing the constant of proportionality equal to 2.0 with the stagnation pressure coefficient behind the normal detached shock wave. In terms of the pressure ratios, the modified Newtonian relationships become:

$$\frac{P}{P_o} = \frac{P}{P_o} + \left(1 - \frac{P_\infty}{P_o}\right)\sin^2\theta \tag{3}$$

where

$P$ = local static pressure on surface,

## TABLE II - CHARACTERISTICS OF VM ATMOSPHERES

| Property | Symbol | Dimension | Atmosphere profile | | | |
|---|---|---|---|---|---|---|
| | | | VM3 | VM4 | VM7 | VM8 |
| Surface pressure | $P_o$ | mb | 10.0 | 10.0 | 5.0 | 5 0 |
| | | lb/sq ft | 20.9 | 20.9 | 10.4 | 10.4 |
| Surface density | $\rho_o$ | (gm/cu cm)$10^5$ | 1.365 | 2.57 | 0.68 | 1.32 |
| | | (slugs/cu ft)$10^5$ | 2.65 | 4.98 | 1.32 | 2.56 |
| Surface temperature | $T_o$ | K | 275 | 200 | 275 | 200 |
| | | R | 495 | 360 | 495 | 360 |
| Stratospheric temperature | $T_s$ | K | 200 | 100 | 200 | 100 |
| | | R | 360 | 180 | 360 | 180 |
| Acceleration of gravity at surface | g | cm/sec$^2$ | 375 | 375 | 375 | 375 |
| | | ft/sec$^2$ | 12.3 | 12.3 | 12.3 | 12.3 |
| Composition | | | | | | |
| $CO_2$ (by mass) | | | 28.2 | 70.0 | 28.2 | 100.0 |
| $CO_2$ (by volume) | | | 20.0 | 68.0 | 20.0 | 100.0 |
| $N_2$ (by mass) | | | 71.8 | 0.0 | 71.8 | 0.0 |
| $N_2$ (by volume) | | | 80.0 | 0.0 | 80.0 | 0.0 |
| A (by mass) | | | 0.0 | 30.0 | 0.0 | 0.0 |
| A (by volume) | | | 0.0 | 32.0 | 0.0 | 0.0 |
| Molecular weight | M | mol$^{-1}$ | 31.2 | 42.7 | 31.2 | 44.0 |
| Specific heat of mixture | $C_p$ | cal/gm C | 0.230 | 0.153 | 0.230 | 0.166 |
| Specific heat ratio | $\gamma$ | | 1.38 | 1.43 | 1.38 | 1.37 |
| Adiabatic lapse rate | $\Gamma$ | K/km | -3.88 | -5.85 | -3.88 | -5.39 |
| | | R/1000 ft | -2.13 | -3.21 | -2.13 | -2.96 |
| Tropopause altitude | $h_T$ | km | 19.3 | 17.1 | 19.3 | 18.6 |
| | | kilo ft | 63.3 | 56.1 | 63.3 | 61.0 |
| Inverse scale height (stratosphere) | $\beta$ | km$^{-1}$ | 0.0705 | 0.193 | 0.0705 | 0.199 |
| | | ft$^{-1}$ x $10^5$ | 2.15 | 5.89 | 2.15 | 6.07 |
| Continuous surface wind speed | v | ft/sec | 155.5 | 155.5 | 220.0 | 220.0 |
| Peak surface wind speed | $v_{max}$ | ft/sec | 390.0 | 390.0 | 556.0 | 556.0 |
| Design vertical wind gradient | dv/dh | ft/sec/1000 ft | 2 | 2 | 2 | 2 |
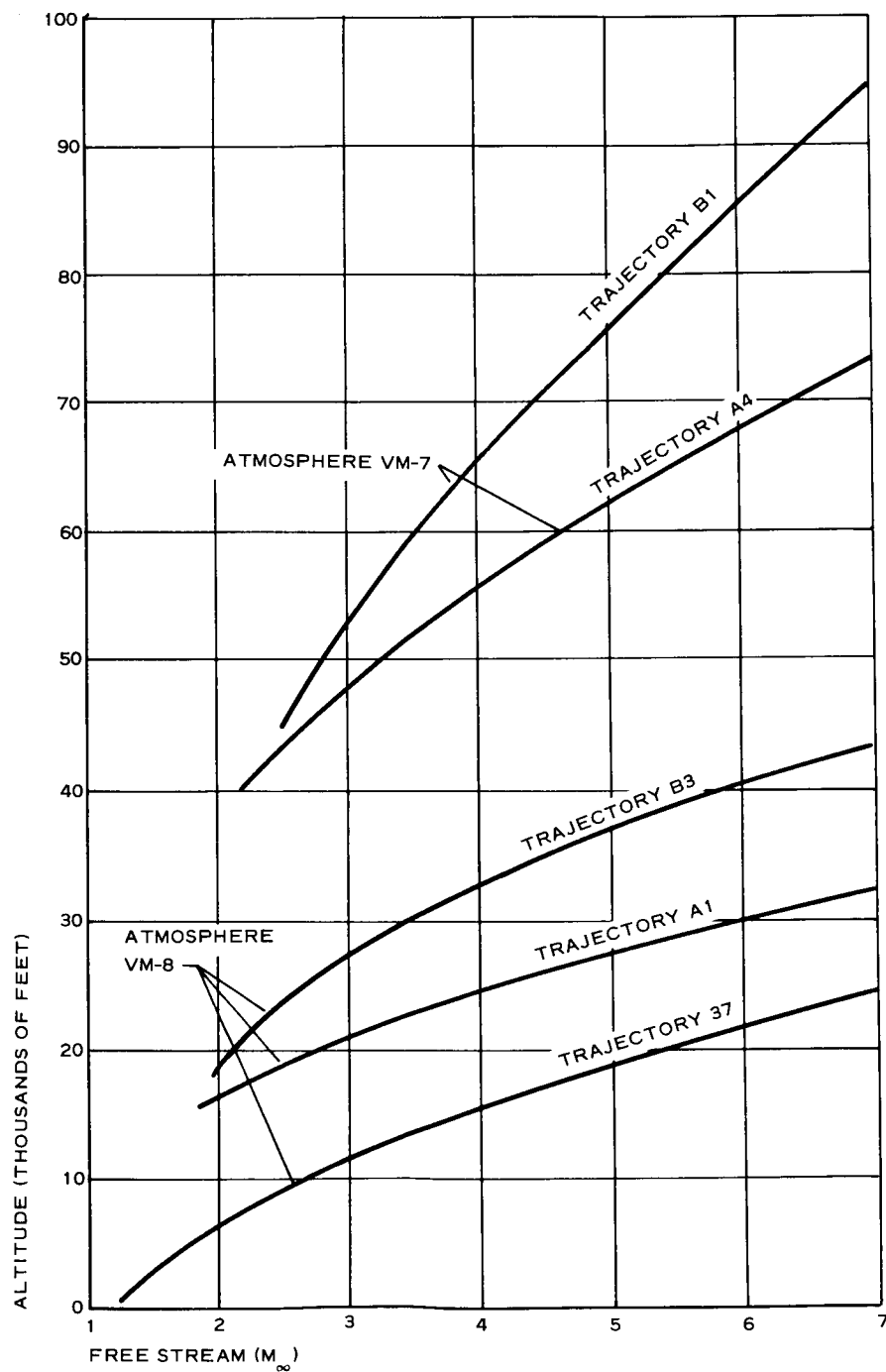
Figure 18 - Initial Mars Entry Trajectories
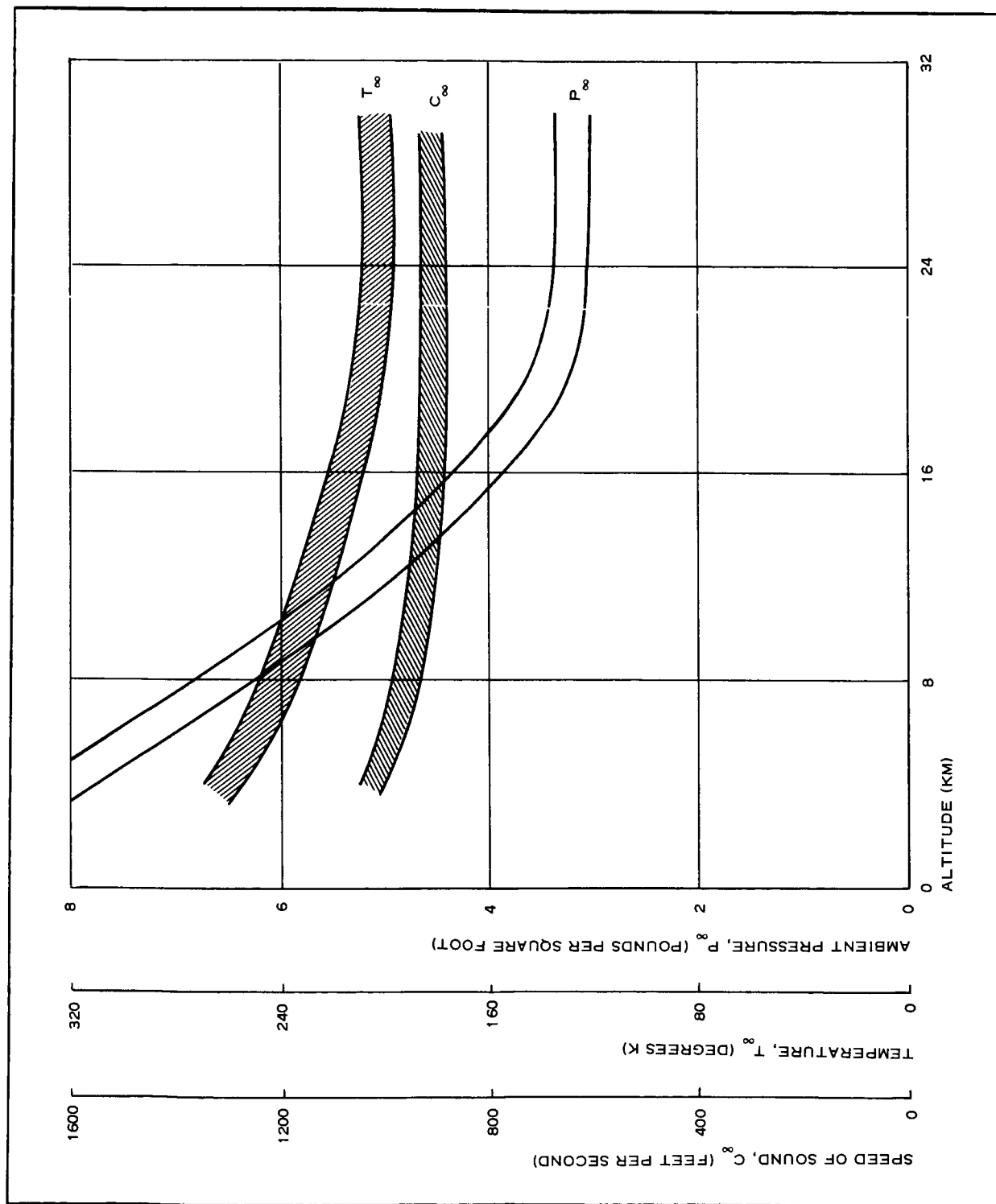
Figure 19 - Decelerator Deployment Envelope, VM7 Atmosphere
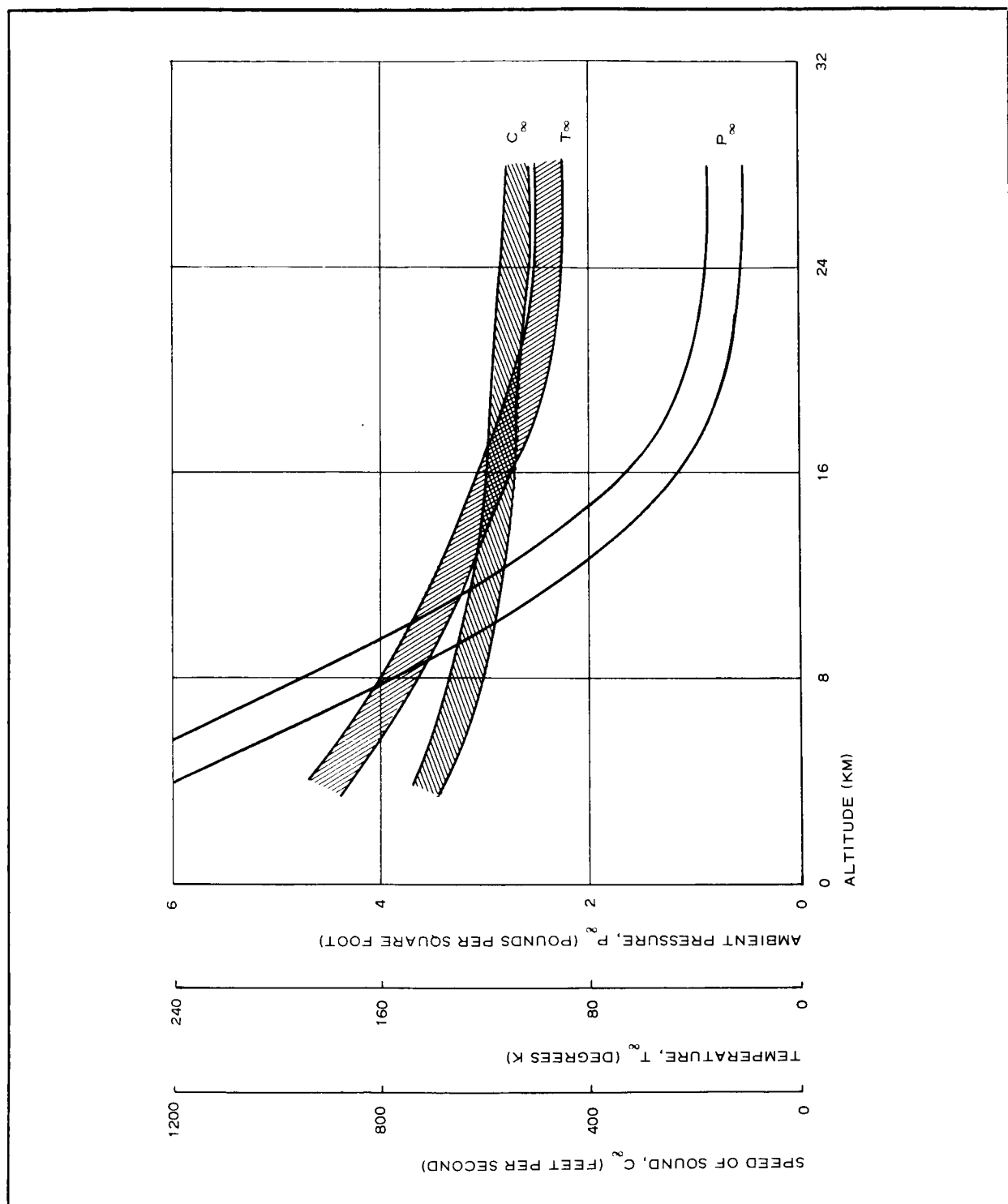
Figure 20 - Decelerator Deployment Envelope, VM8 Atmosphere

$P_o$ = stagnation pressure behind normal shock,

$P_\infty$ = free-stream static pressure, and

$\theta$ = local-surface inclination angle.

This approach has limitations, such as:

1.   Existence and condition uniqueness require high supersonic flows.

2.   Leading edge of the configuration should be blunted.

3.   Accuracy decreases as surface slope with respect to stream decreases.

4.   The effective value of specific heat ($\gamma$) changes significantly across the normal shock in equilibrium real-gas hypersonic flow, but it remains about constant over the leading edge of the blunt body downstream of the stagnation point.

The burble fence of a BALLUTE[a] causes flow separation at the fence, a phenomenon similar to the presence of a forward-facing step in supersonic flow. This phenomenon is associated with a pressure plateau with its peak located at the fence and with the magnitude proportional to the local Mach number and the turning angle of the flow. The separation shock is assumed to be negligible with no local total pressure loss resulting from it.

For capsule-decelerator configurations shown in Figures 21 through 24, pressure distributions on the capsule also are considered because the systems are integrated. The capsule geometry and attached conical flare decelerator are shown in Figure 21. The shapes of this configuration and a BALLUTE up to about $Y/R_D \approx 1.0$, where separation begins because of the fence, are not slender; hence, the blunt body approach is justified for the indicated flow region.

[a]TM, Goodyear Aerospace Corporation, Akron, Ohio.

$R_V$ = 94 IN.

$R_D$ = 1.73 $R_V$

r = 0.09 $R_V$

$r_1$ = 0.0378 $R_V$

$\ell$ = 0.308 $R_V$

$\theta$ = 60 DEG

SHOCK WAVE
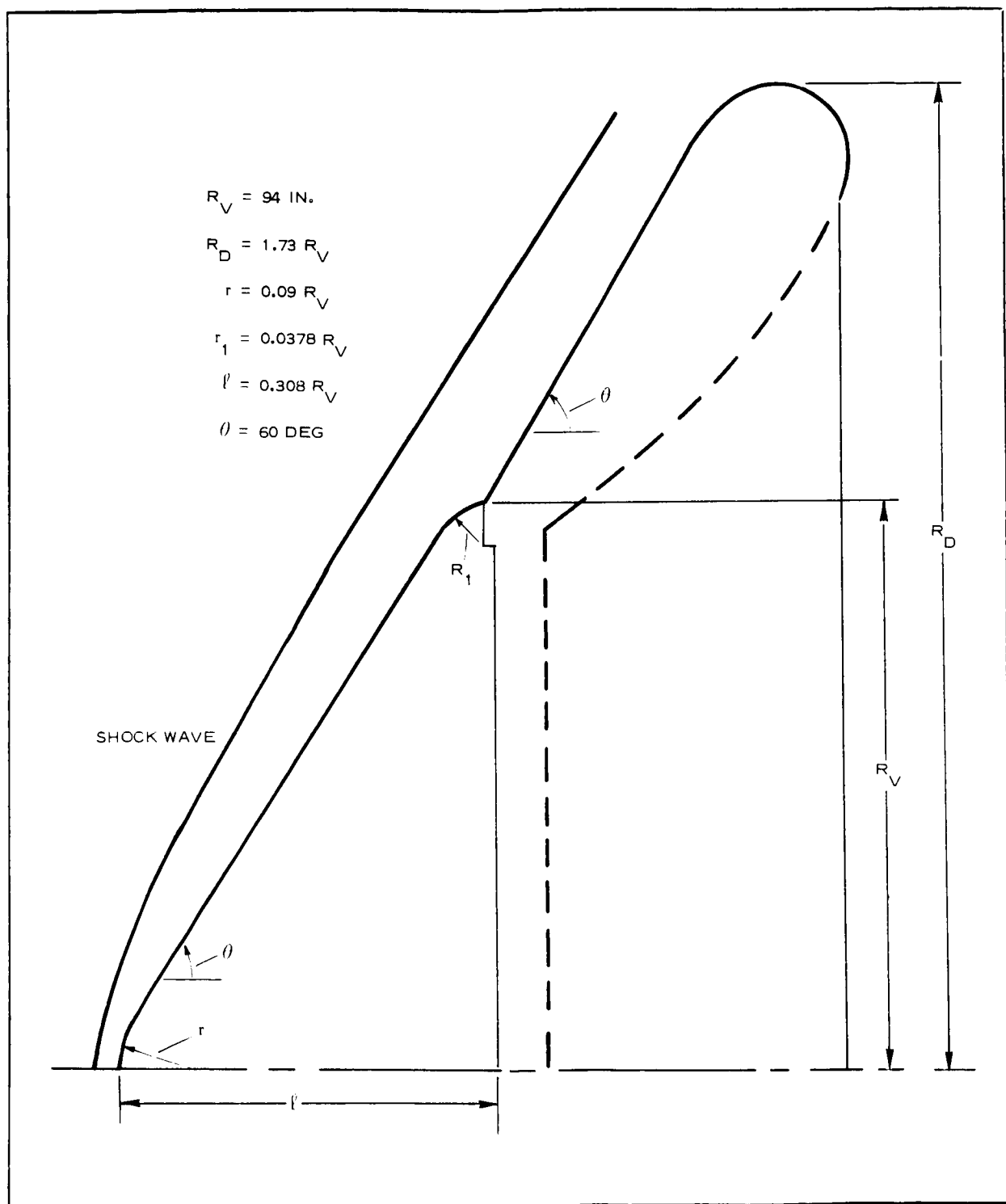
Figure 21 - Entry Capsule and Decelerator Configuration
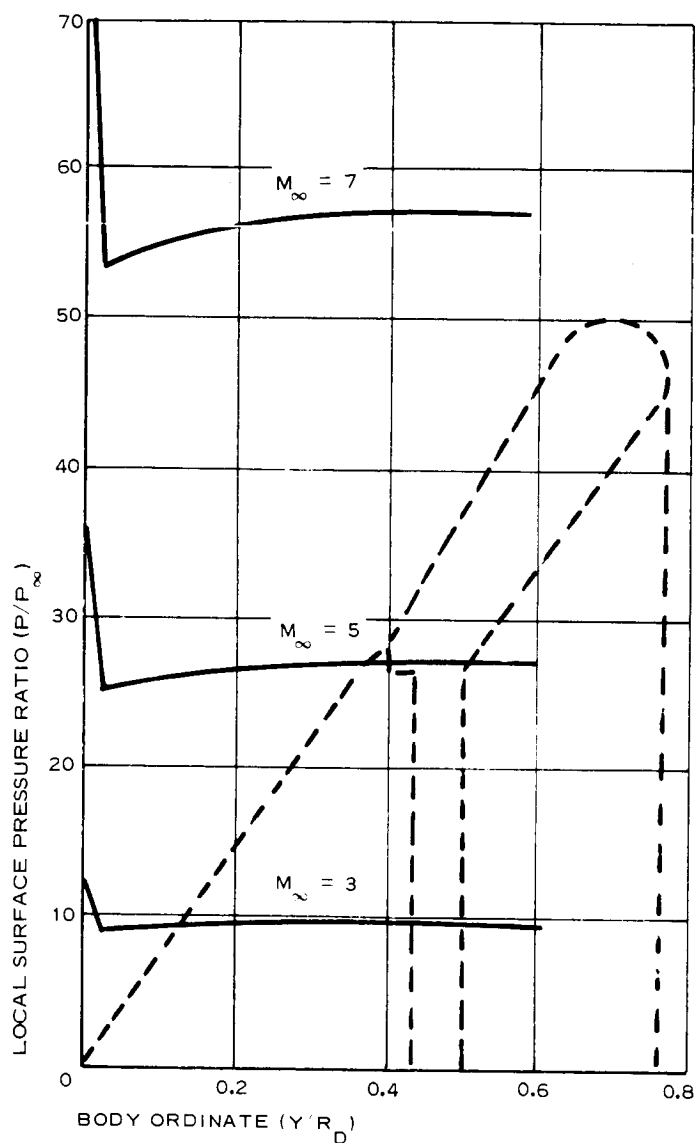
Figure 22 - JPL Capsule-Conical Flare Decelerator,
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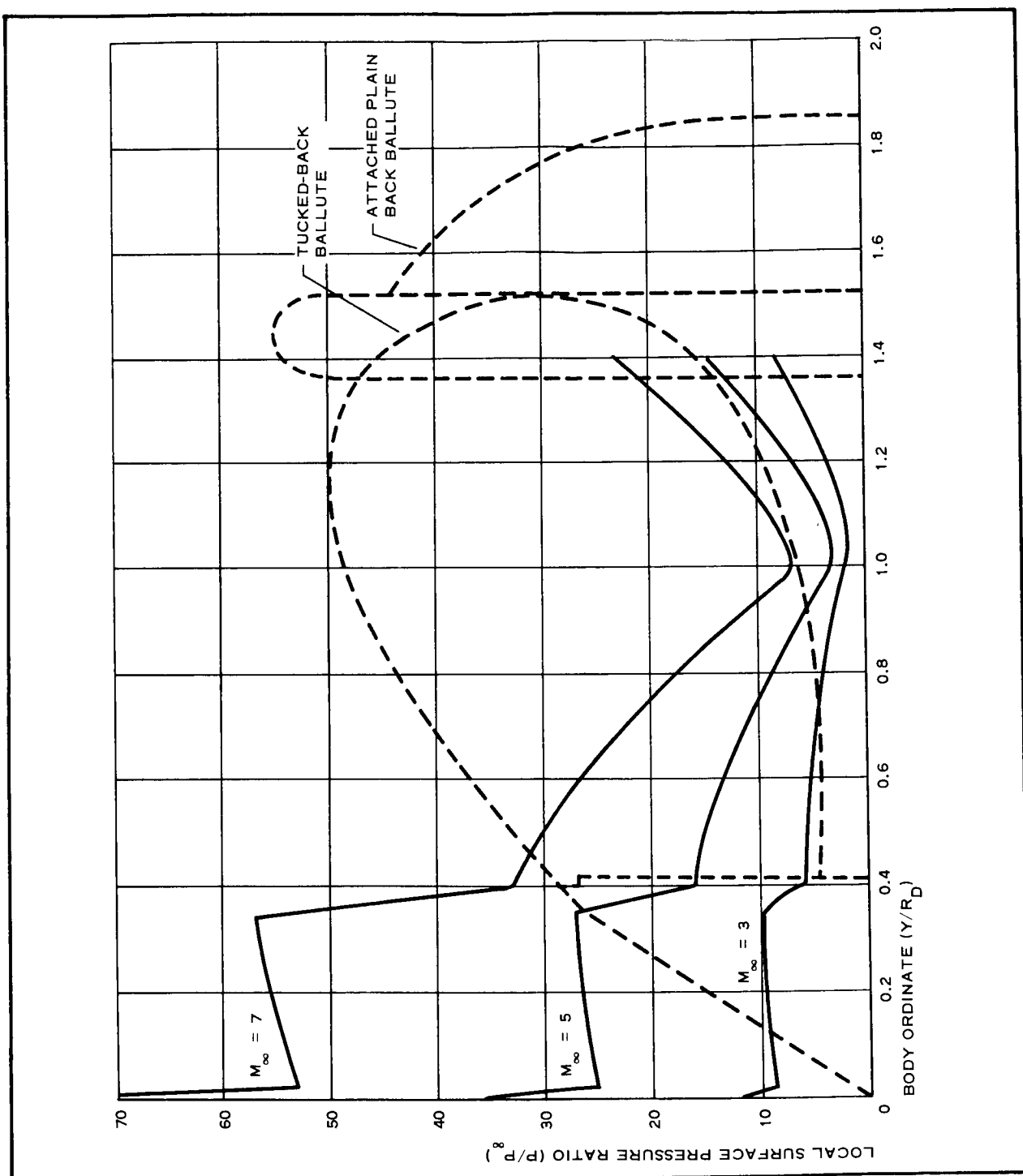
Figure 23 - JPL Capsule-BALLUTE Decelerator
Estimated Pressure Distribution
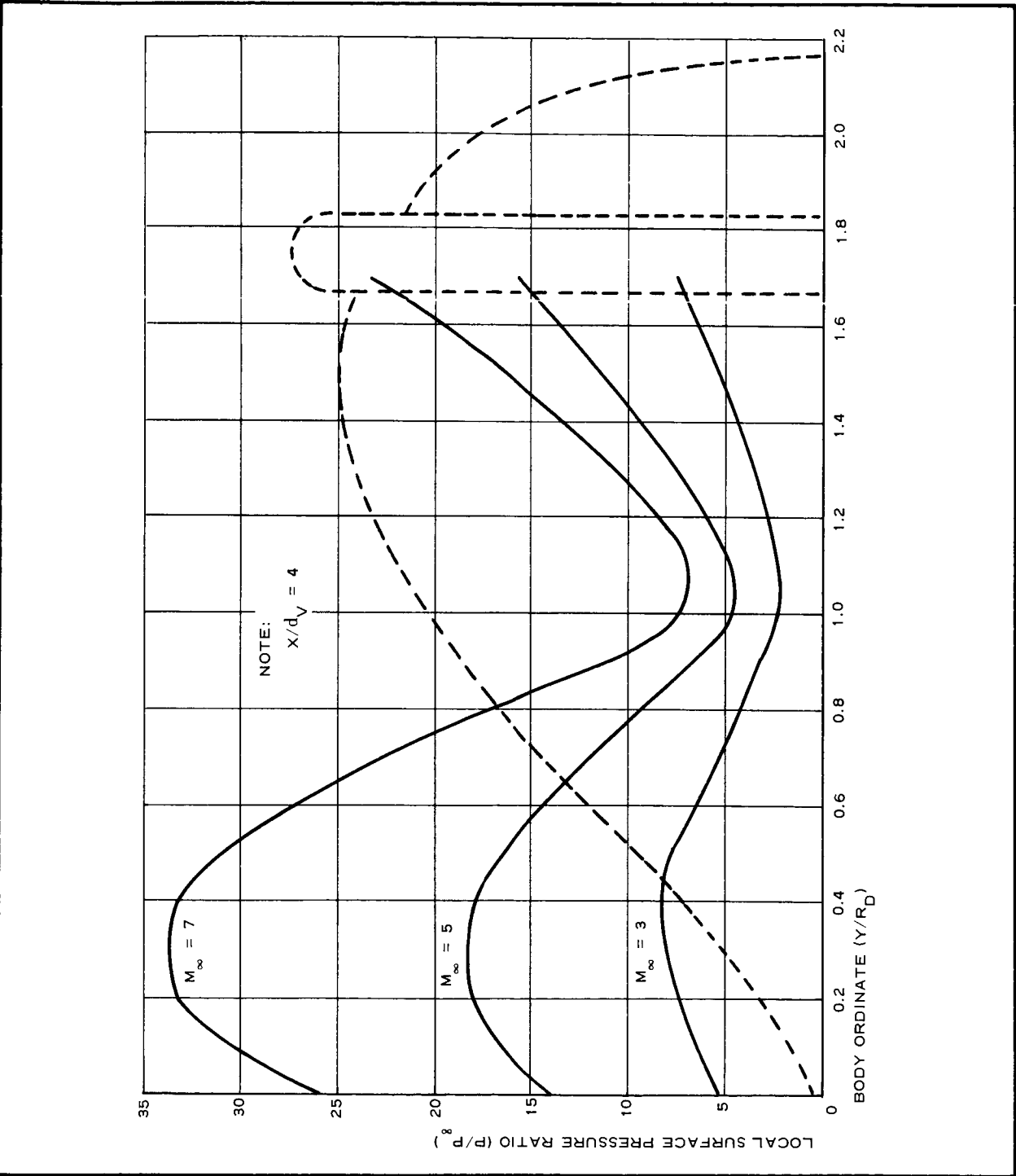
Figure 24 - Trailing BALLUTE Decelerator,
Estimated Pressure Distribution

No direct attempt was made in this study to investigate the shock-wave shape and structure, but a first order approximation for $M_\infty$  3 shock shape is shown in Figure 21.

The resultant estimated pressure distribution over the capsule-conical flare configuration is shown in Figure 22. Similarly, Figure 23 presents an estimated pressure distribution for the capsule-attached BALLUTE configuration. For a BALLUTE trailing the capsule at $X/d_v \equiv 4$, it is necessary to consider the effect of the capsule wake on the flow field of the BALLUTE. According to the available information the viscous wake width at this x/d is a function of the forebody shape and Mach number of the free stream as the principal variables. Thus, the forebody base radius at the flow conditions under consideration is related to wake half-width as follows:

$$0.6 \lesssim \frac{y_w}{R_v} \lesssim 0.8,$$

where

$y_w$ = viscous wake half-width and

$R_v$ = forebody base radius.

Based on wake geometry and on wake velocity "frozen" at the outer edge of the viscous wake at about $M \approx 3.0$ behind blunted bodies, associated pressure distributions are estimated in Figure 24.

Experience shows loading on the after part of a BALLUTE is inversely proportional to $M^2_\infty$. In terms of pressure ratios, estimated values are:

| $M_\infty$ | $P/P_\infty$ |
| --- | --- |
| 3 | 1.570 |
| 5 | 1.650 |
| 7 | 1.685 |

Results of this short inquiry are far from precise and are conservative. Knowledge of the Martian atmosphere is obtained from remote observations and indirect experiments that offer theoretical interpretations based on the physical processes occurring on the Earth.

4.    THERMAL ENVIRONMENT

   a.    Deployment Conditions

      Deployment of a first-stage decelerator for the Mars lander vehicle in the near surface region of the Martian atmosphere was initially established in Mach range $2 < M_D < 7$.

      The approximate severity of aerodynamic heating may be estimated with deployment condition data illustrated in Figures 25 and 26. Data in Figure 25 are for the VM7 model atmosphere. Figure 26 data are for the VM8 model atmosphere. Lines of constant dynamic pressure and adiabatic wall temperature using a recovery factor of 0.9 are shown as functions of altitude and deployment Mach number. In general, deploying the first-stage decelerator at the low end of the Mach number range is not expected to produce significant aerodynamic heating in either atmosphere because of the potentially low thermal environment. However, as the deployment Mach number is increased to five or higher, the prospects of aerodynamic heating are increased. In the VM7 atmosphere, for example, at a deployment speed of Mach 5 and altitude of 65, 000 ft, the adiabatic wall temperature becomes about 1500 F and dynamic pressure 60 psf. At Mach 7, the adiabatic wall temperature increases to about 3, 000 F for deployment at 85, 000 ft and approximately the same dynamic pressure.

      With similar deployment conditions in the VM8 atmosphere, aerodynamic heating prospects encountered are approximately the same as for the VM7 model atmosphere. Thus, heating in the two model atmospheres will be about the same.
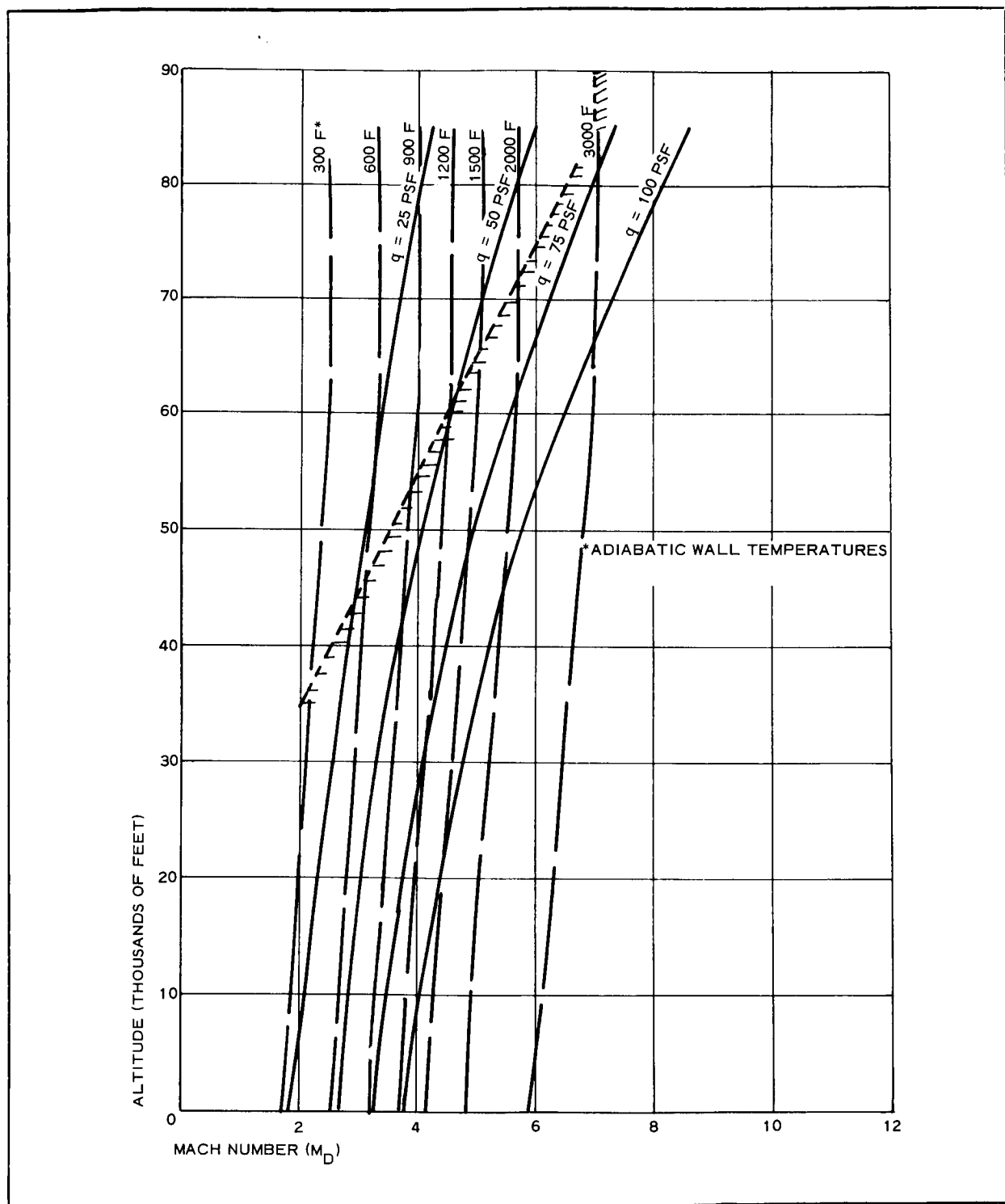
Figure 25 - Deployment Conditions, VM7 Atmosphere
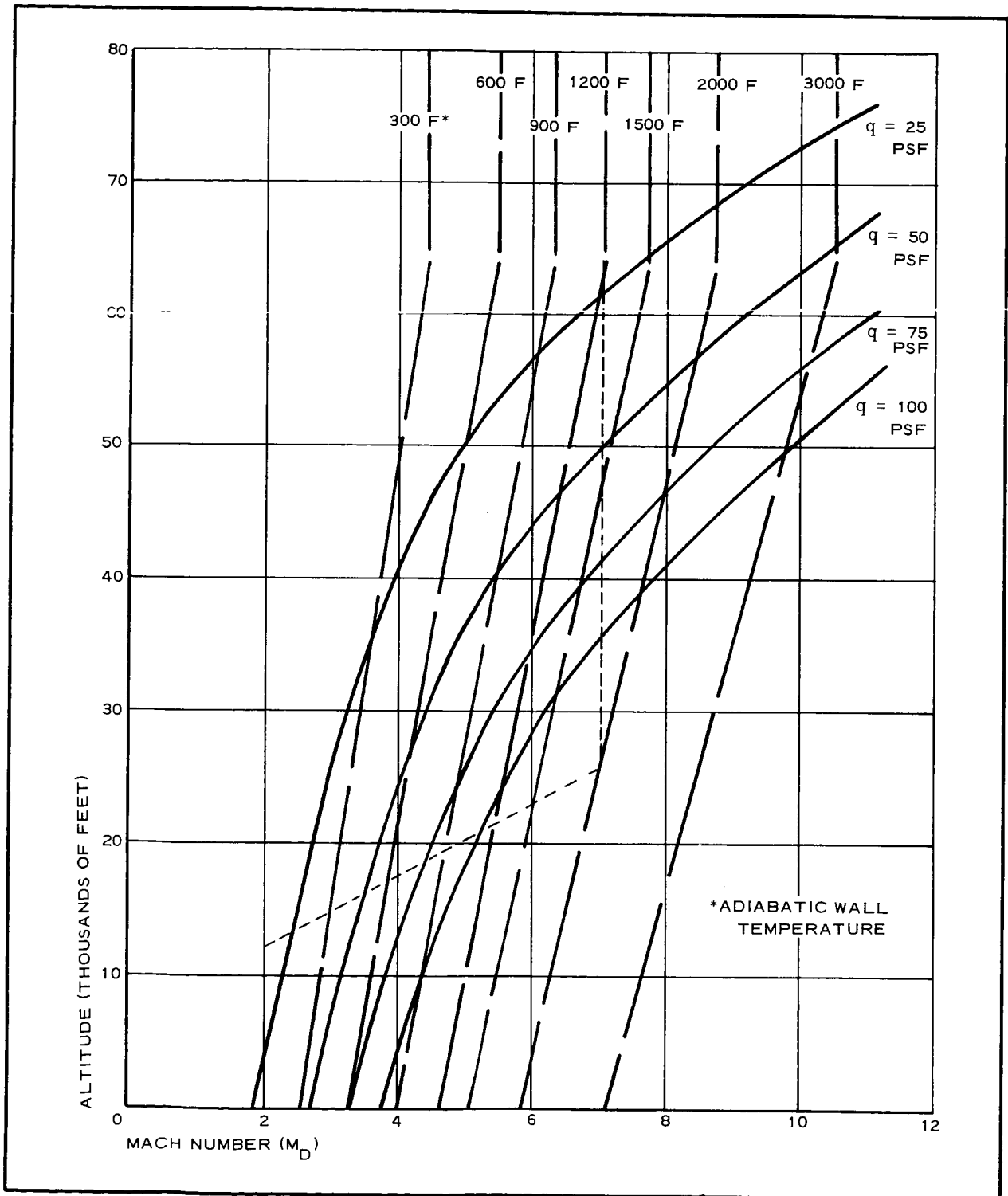
Figure 26 - Deployment Conditions, VM8 Atmosphere

Figure 25 and 26 show limits of the deployment envelope established from trajectory analysis to attain final-stage decelerator deployment conditions of about Mach 1 above 10,000 ft. It appears that the severest deployment cases ($5 < M_D < 7$) will encounter adiabatic wall temperature of about 3000 F and maximum dynamic pressures of 100 psf. Because these conditions are generated at the upper end of the deployment regime, these flight conditions must be examined more rigorously to determine the effect of aerodynamic heating on decelerator material.

b.  Heat Flux Rates

The first-stage decelerators studied in this program were (1) attached conical flare, (2) attached plain-back BALLUTE, (3) tucked-back BALLUTE, (4) trailing BALLUTE. The single position directly amenable to heat-flux rate analysis for atmospheres other than air is the stagnation point of entry capsules. In particular,

$$\dot{q}_o = \bar{C}\left(\frac{P_o}{R_N}\right)^{1/2} (U_\infty)^N (1 - g_w) , \tag{4}$$

where

$\dot{q}_o$ = Stagnation point heat flux rate, $BTU/ft^2$-sec;

$P_o$ = Stagnation point pressure, atmospheres;

$R_N$ = Nose radius (feet);

$U_\infty$ = Free-stream velocity divided by 10,000 fps;

$g_w$ = Ratio of wall to total enthalpy;

$\bar{C}$ = Constant depending on gas composition; and

$N$ = Exponent depending on gas composition. [1,a]

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[a]Superior numbers in the text refer to items in the List of References.

This equation is well suited for estimating the stagnation point heat-flux rate during the deceleration profile. Thus, it appears appropriate first to generate this heating rate and then to estimate the heating rates further back on the body on the basis of the stagnation point heat-flux rates. Typical stagnation point heat-flux rates are shown in Figure 27 for VM7 and VM8 atmospheres, as functions of flight velocity. These rates depend on stagnation pressure, total enthalpy, and body geometry. After the trajectory has been specified, rates may be estimated from flow properties generated at the stagnation point as a function of velocity and altitude.

Heat-flux rates to the surface of interest (that is, the expandable flares or BALLUTES) then can be calculated from the generated stagnation-point heat flux rates. Simplest is Lee's distribution method.[2] Assuming a hemispherical blunt-nose body over which laminar flow exists on the forebody surface and assuming also a 60-deg half angle
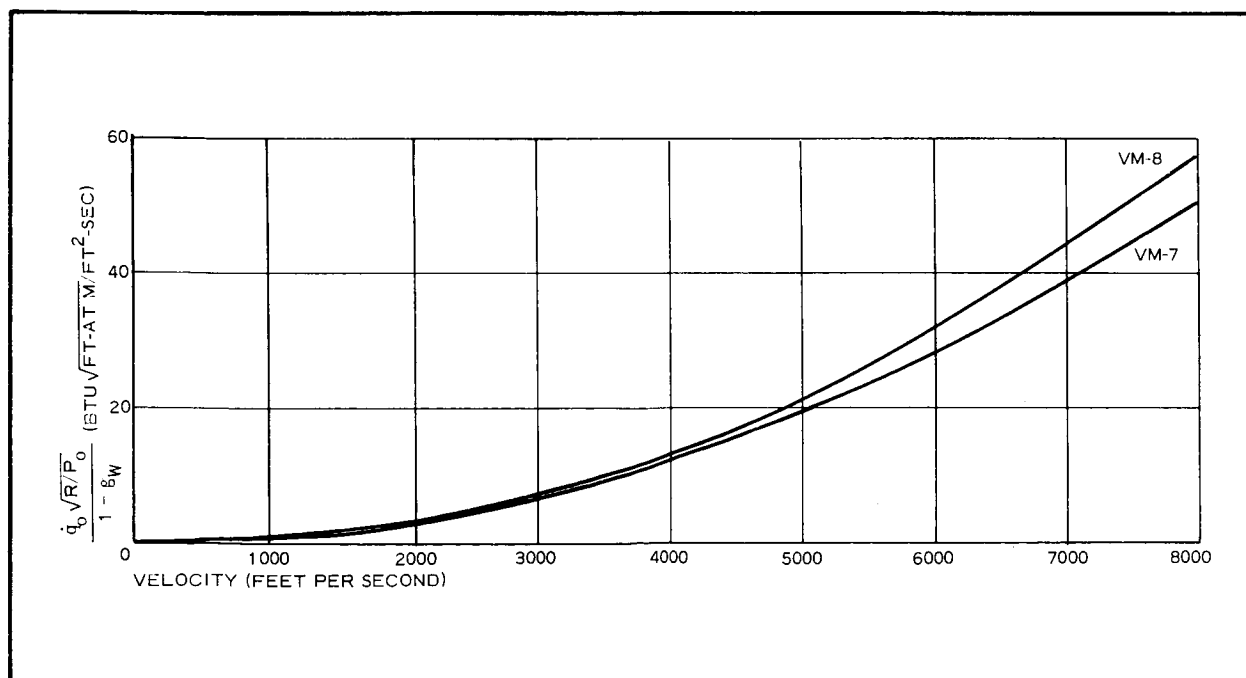


Figure 27 - Stagnation Point Heat-Flux Rate

with an expandable flare of similar geometry, the ratio of local heat-flux rate to stagnation point heat flux can be calculated using Lees' equation.

On the basis of a modified Newtonian pressure distribution over this forebody surface, an evaluation of the local heat-flux rate showed it to be approximately 25 percent of the stagnation-point rate. To confirm this estimated heat flux rate magnitude, Goodyear Aerospace surveyed reports of experimental heat-transfer data dealing with the testing of blunt face configurations somewhat similar to the present configurations. Data obtained are shown in Figure 28 as functions of unit Reynolds number. Single-point data for a hemispherical nose followed by half-angle flares of 30, 50, and 70 deg were obtained.[3] These data showed that the local heat flux rate averages about 20 to 40 percent of the stagnation point heat-flux rate. The local heat-flux rate variation for the 40-deg half angle configuration was obtained.[4] The upper curve was obtained for a solitary BALLUTE and the lower curve for a BALLUTE trailing in the wake of a leading body. The variation in local heat flux rate was again about 20 to 40 percent of the stagnation point heat-flux rate except with Reynolds numbers above $10^6$ where rates begin to rise significantly.

Because of this cursory examination of local heat-flux rate distribution for various configurations, Goodyear Aerospace assumed that a local heat-flux rate of 0.3 of the stagnation-point heat-flux rate would be a representative value for estimating heat-flux rate profiles during deceleration of all configurations. Heat-flux rate profiles for various trajectories and atmospheres are shown in Figures 29 through 35 for a wall temperature of 70 F. These heat-flux rate profiles show the effect of increasing drag potential of the decelerator device. In terms of the heat-flux profile, because deployment conditions are identical, increasing decelerator drag potential decreases heating exposure time.
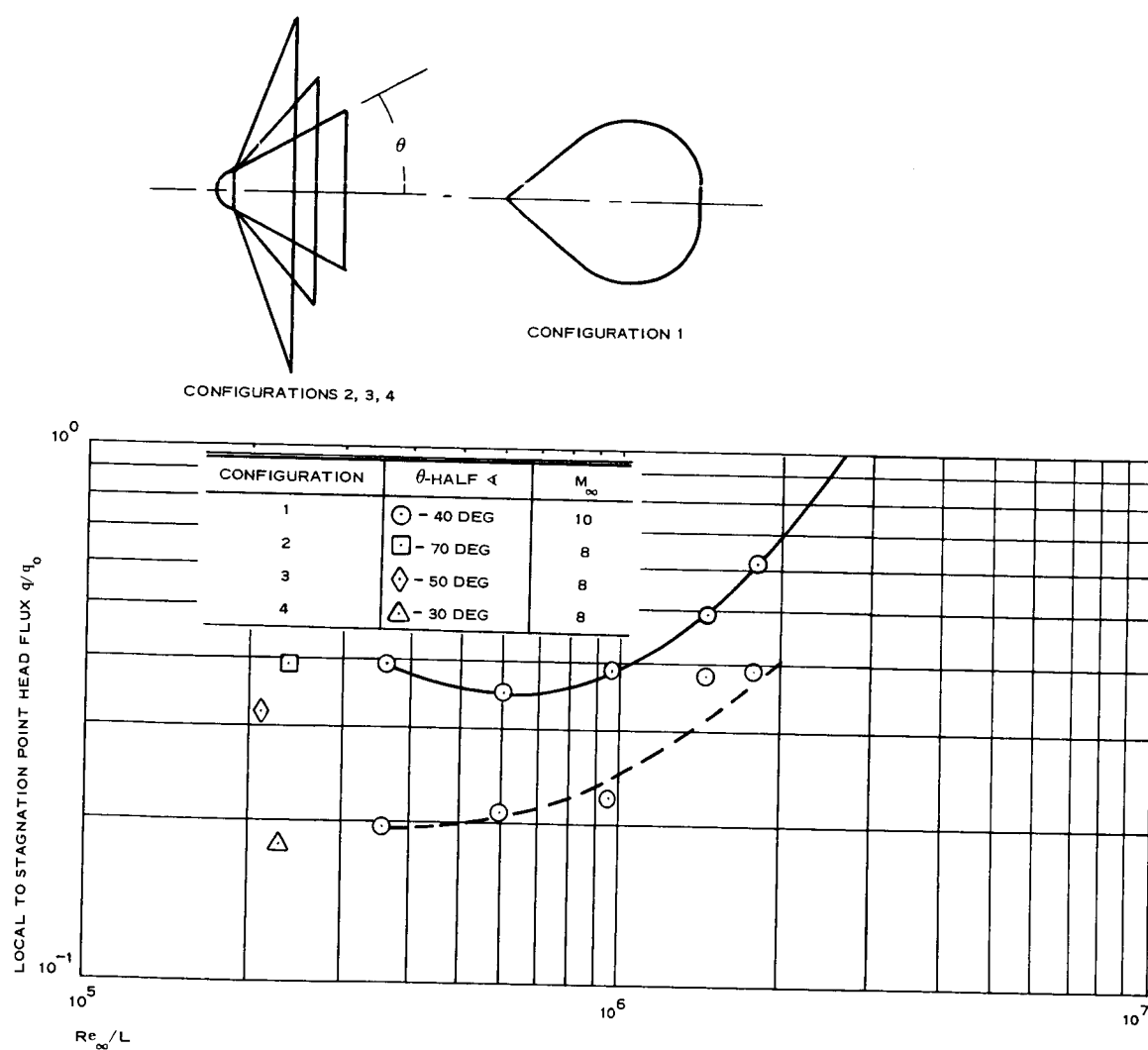
Figure 28 - Local Heat-Flux Rate versus Unit Reynolds Number
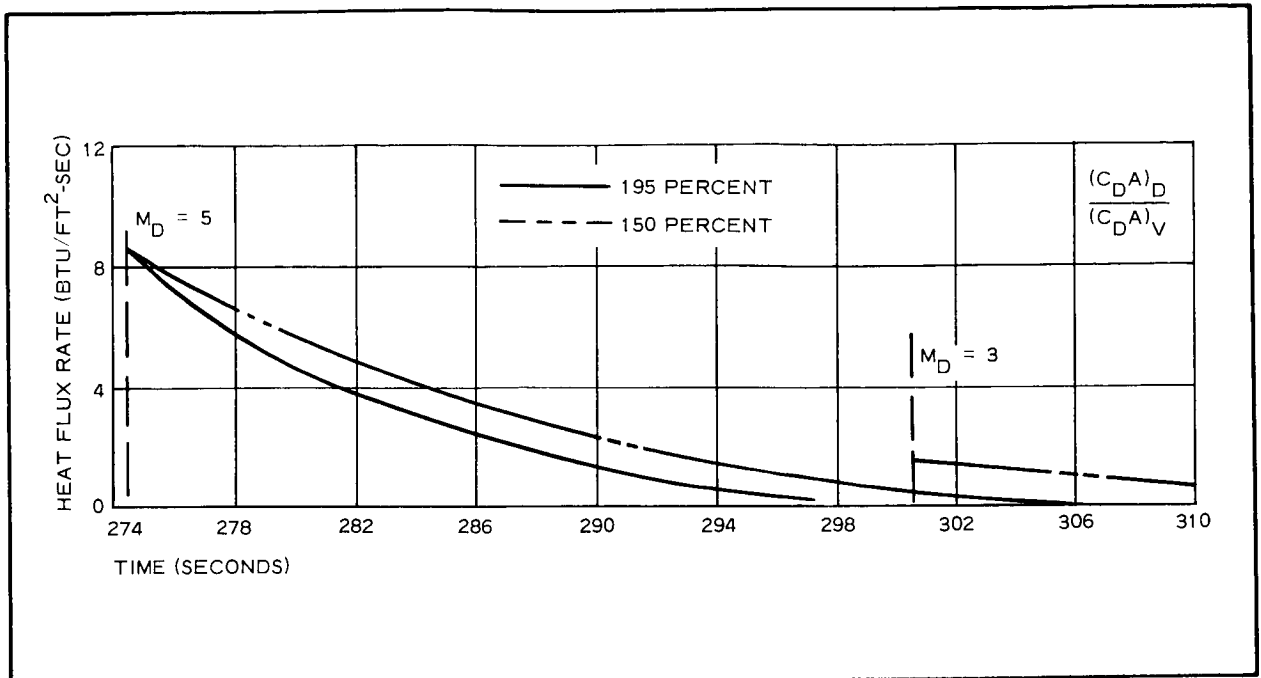
Figure 29 - Heat-Flux Rate, Trajectory B1
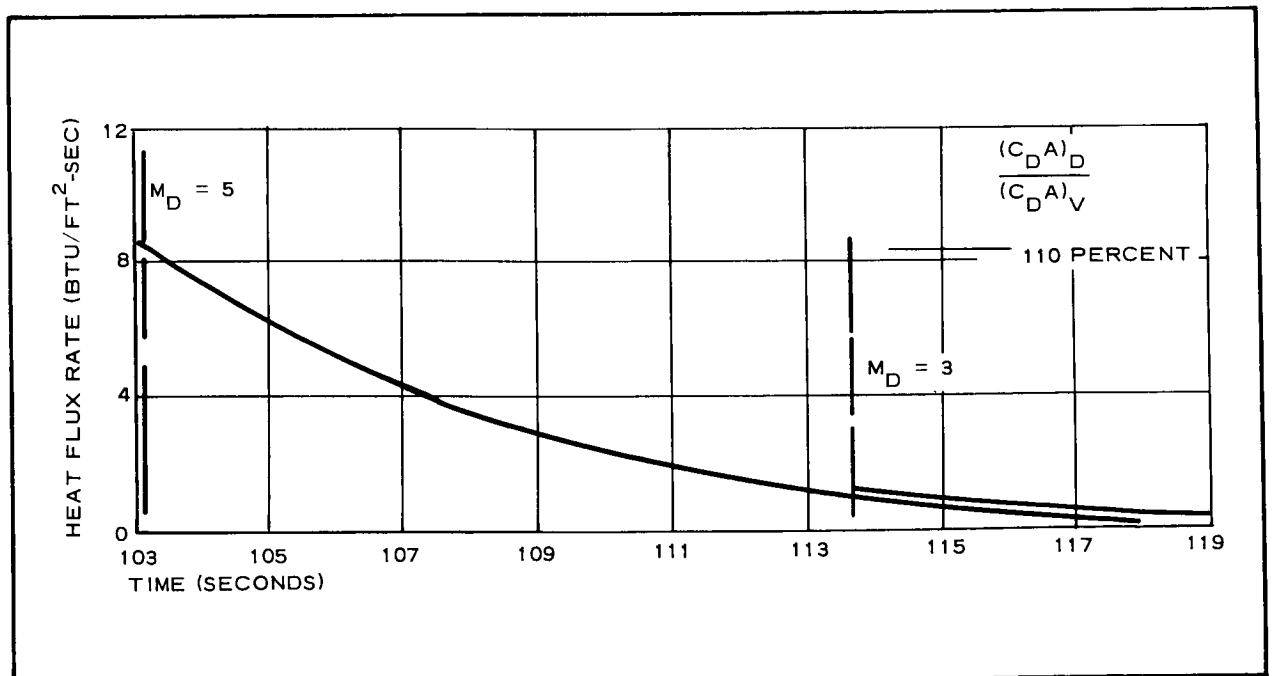


Figure 30 - Heat-Flux Rate, Trajectory A4
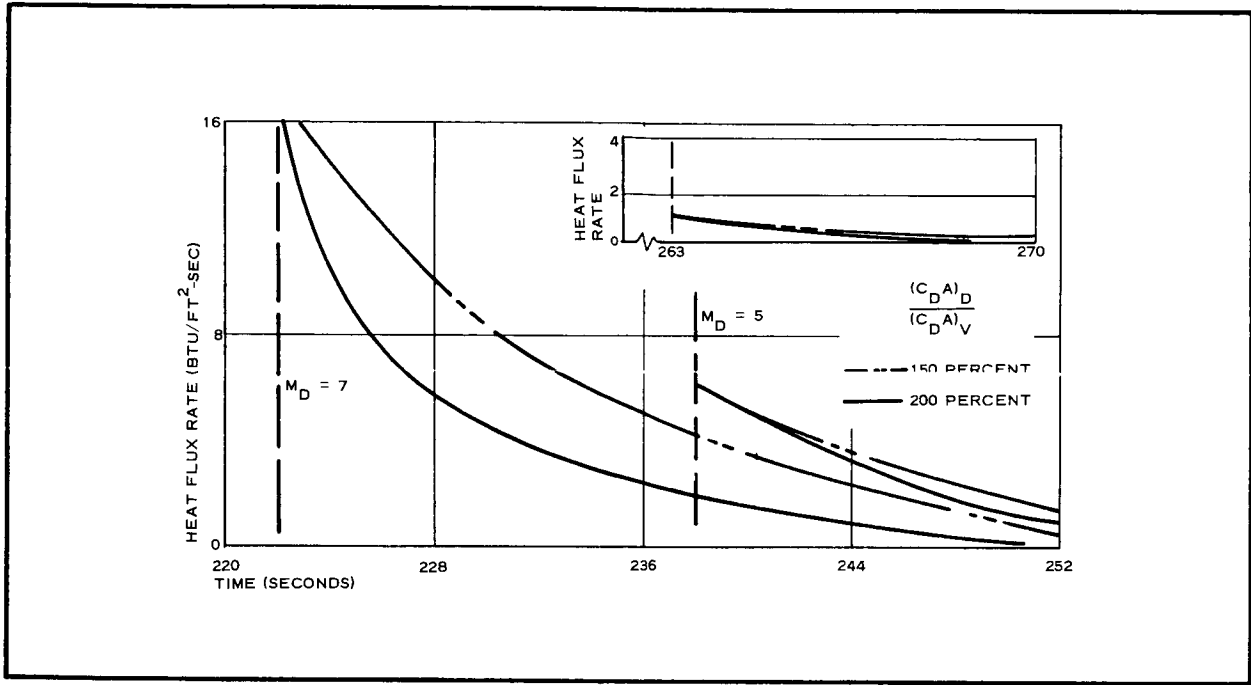
Figure 31 - Heat-Flux Rate, Trajectory 22
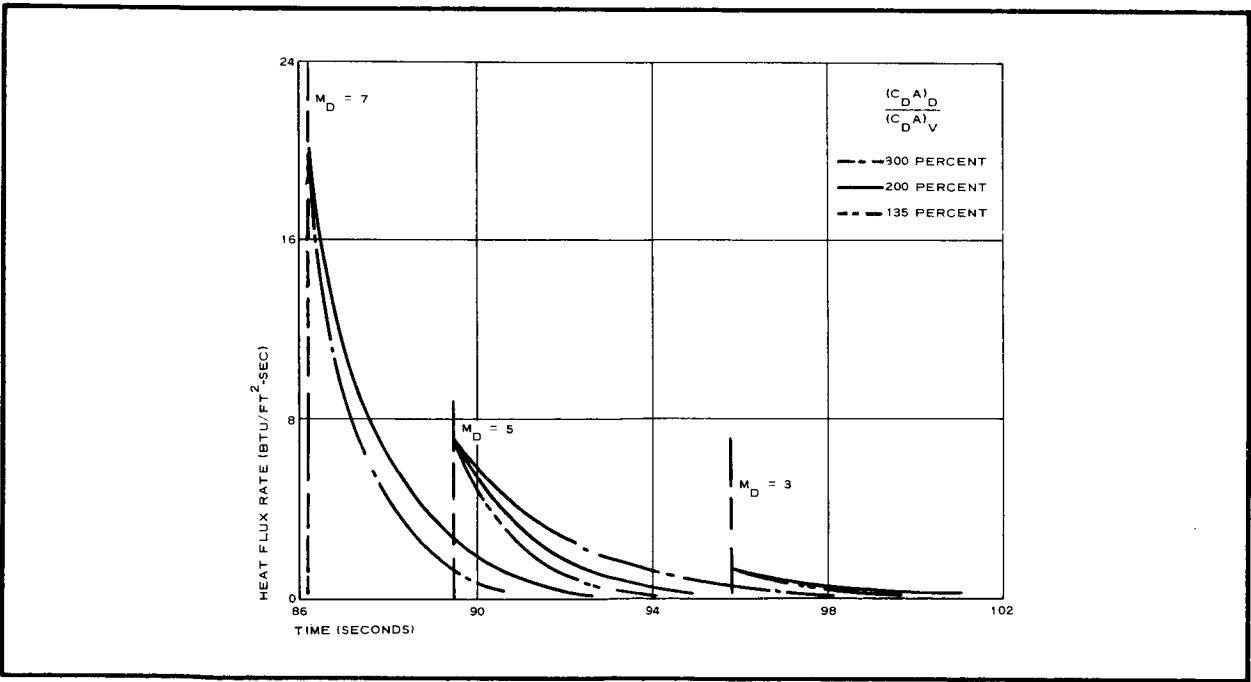


Figure 32 - Heat-Flux Rate, Trajectory 19
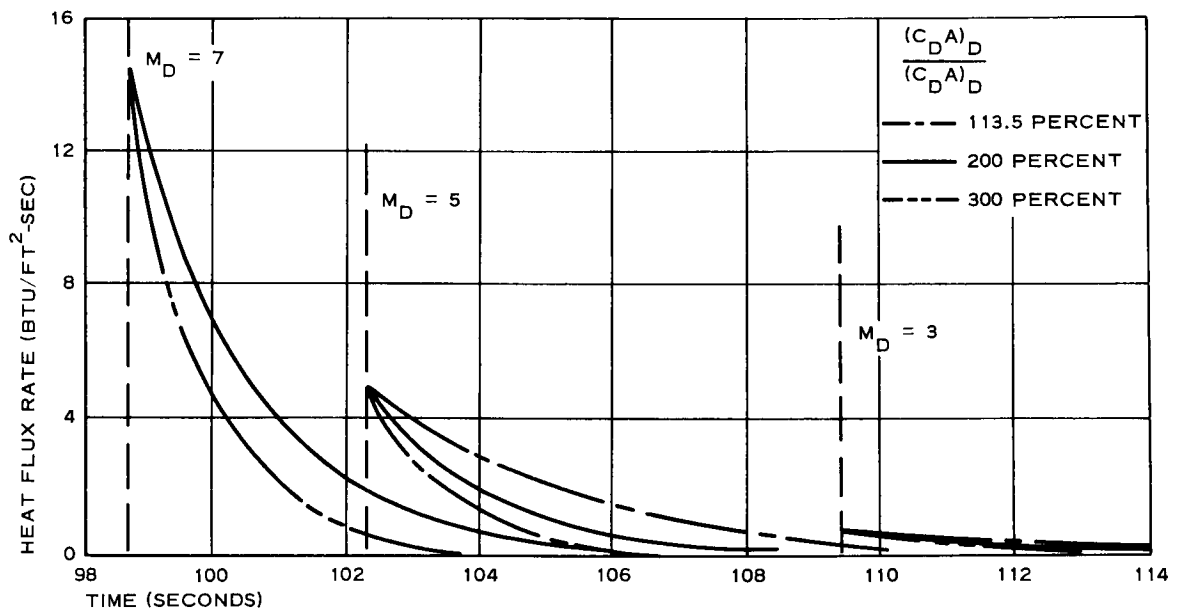
Figure 33 - Heat-Flux Rate, Trajectory 37
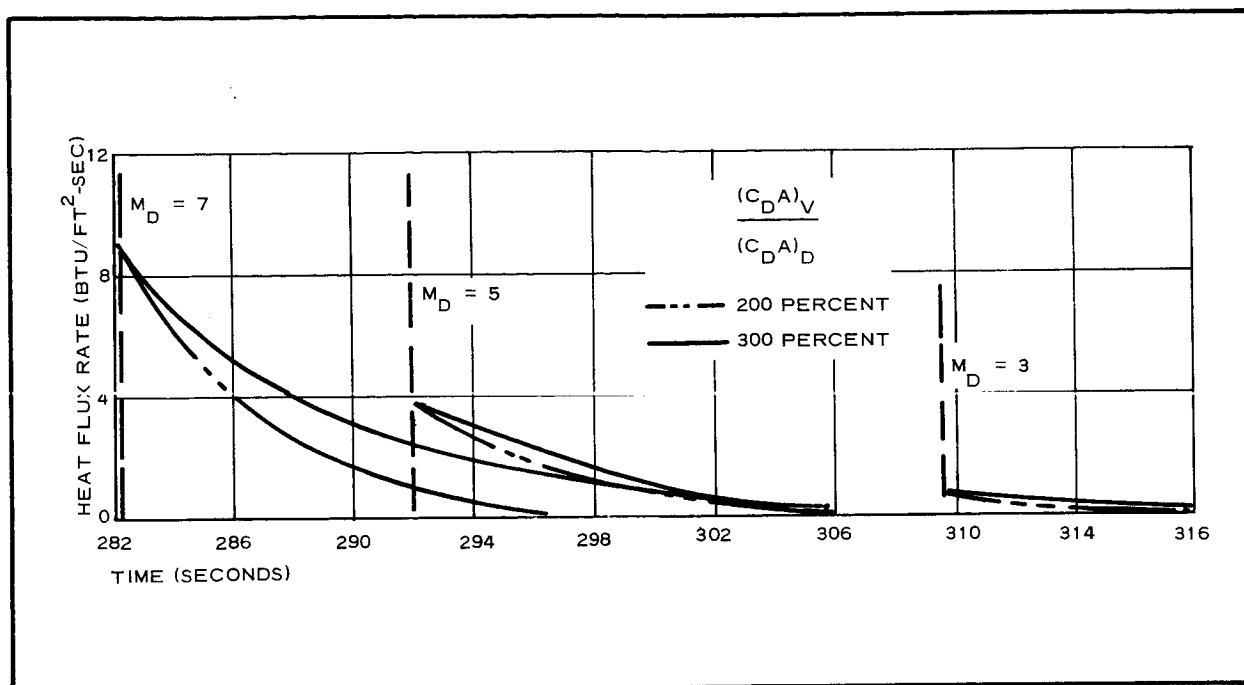
Figure 34 - Heat-Flux Rate, Trajectory A1
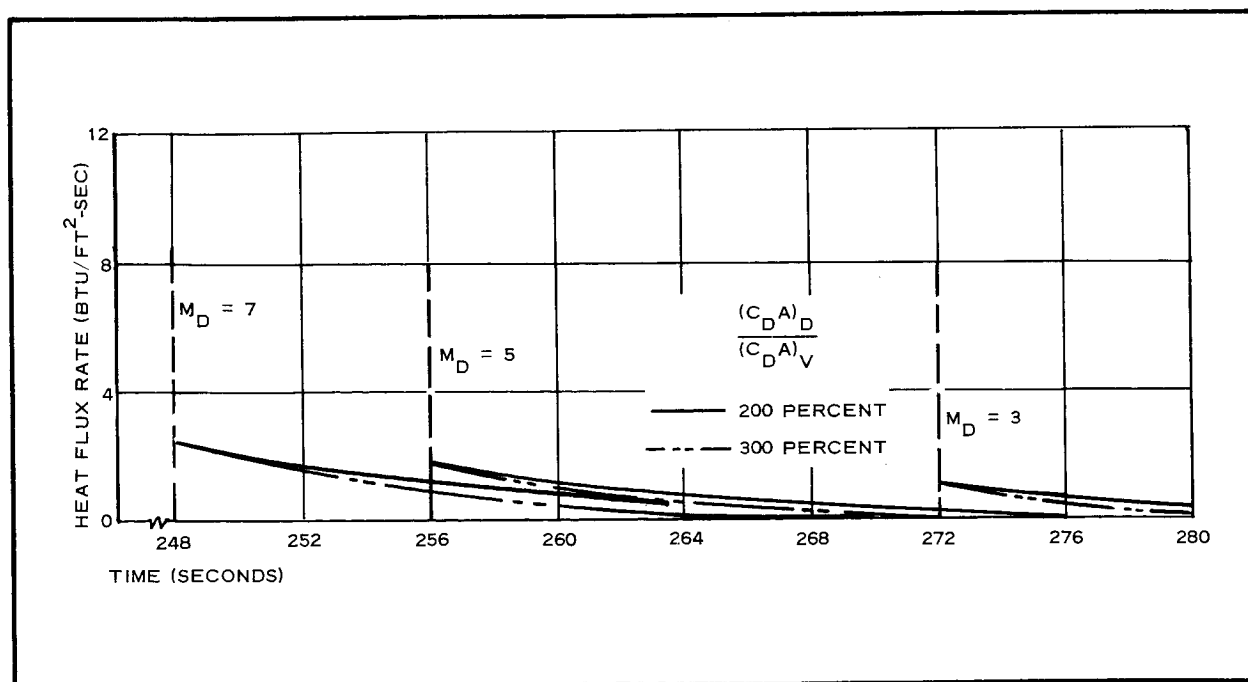


Figure 35 - Heat-Flux Rate, Trajectory B3

Maximum heat-flux rates encountered occur immediately at deploy-
ment and appear to be well within the capability of the fabrics. How-
ever, deployment at Mach 7 in trajectory 22 of the VM7 atmosphere
and trajectory 37 of the VM8 atmosphere indicate that the initial
heat-flux rates may be quite substantial, although time of decelera-
tion heating is quite short.

## SECTION III - STERILIZATION AND MATERIALS

1. STERILIZATION REQUIREMENTS AFFECTING DESIGN

    a. General

    Planning to meet sterilization requirements for the Mars entry cap-
    sule requires these guide lines:

    1. Sterilization techniques

    2. Spacecraft design for sterilization

    3. Sterile assembly techniques

    4. Compatibility of materials and components
       with sterilization procedures

    b. Sterilization Techniques

    Dry heat is recommended for sterilization in most cases. JPL has
    established sterilization specifications for Type Approval and Flight
    Acceptance equipment for the Mars entry capsule.[5] If the entry cap-
    sule could be encapsulated in a biological barrier with proper heat
    soak and kept within this barrier until just before landing, the steri-
    lization requirement would be met. Voyager entry capsule design
    plans are proceeding with this technique.

    Because of two factors, other sterilization techniques are being in-
    vestigated, however. An entry capsule capable of surviving the heat
    soak may not be possible. If this is true, parts that cannot survive
    must be sterilized differently and inserted with sterile techniques
    into the capsule after heat soak. The second factor is the time re-
    lationship of various sterilization processes. As the number of
    organisms increases, time required to reach a given level of sterility

increases. A knowledge of initial contamination is required to determine final condition.

Amont toxic chemicals that can be used in a gaseous state, ethylene oxide generally is preferred for surface sterilization of spacecraft. Currently, a blend of 12 percent ethylene oxide and 88 percent Freon 12 is suitable for spacecraft sterilization. Other compounds are available for special applications. Among these are formaldehyde, methyl bromide, betapropialactane, ethylenimine, and paracetic acid. Available wipe-on disinfectants include plenol, quaternary bases, mustards, and organic solvents.

Radiation effectively sterilizes. Sufficient doses of ultraviolet below 3200 A provide surface sterilization; however, penetration is poor. X-rays and gamma rays have greater penetrating powers, but these are detrimental to many spacecraft components.

Filtration is available for sterilizing liquids and gases. The smallest organisms known are from 0.01 to 0.1$\mu$ . Filters with small pore size combined with a high dielectric constant across the filter used in a series can provide statistical assurance of sterility.

c.  Spacecraft Design for Sterilization

Various research efforts have sought to determine compatibility of materials and components with sterilization methods and environments. The JPL approach to the entry capsule is to develop a qualified parts list for dry-heat sterilization and to design the capsule using this list. Design-around techniques will be used and other parts will be sterilized for insertion after heat soak in the capsule.

As a backup method for heat sterilizing the entire capsule, individual parts can be sterilized, assembled in a sterile environment, and inserted into the spacecraft after the major part of the spacecraft has been heat sterilized. Clean-room techniques with added biological considerations would be used.

Placing a sterile spacecraft or entry capsule on a planetary surface requires consideration of sterilization techniques from beginning of design. Sterilization must be integrated into each design phase.

These design concepts should include:

1.  Minimum assembly contact points, voids in structure, and assembly steps

2.  Maximum use of self-sterilizing components and materials or those rendered sterile during manufacture

3.  Maximum assembly of the spacecraft prior to internal sterilization by dry heat.

4.  Sterile assembly of the major subsystem, terminal surface sterilization, and emergency subsystem replacement or resterilization

5.  Retention of payload encapsulation until an established distance from earth is reached.

A subsystem that must be designed from the beginning for sterilization is the spacecraft structure. Sharp edges and crevices must be held to a minimum to prevent collection of organisms. The maximum surface exposure must be attained for ease of surface sterilization. Efficient heat flow and dimensional stability must be obtained for ease of heat sterilization. Overall geometry must remain simple for ease of encapsulation.

d.  Sterile Assembly of Spacecraft

Basic techniques of sterile assembly, derived from techniques developed in biological laboratories, are:

1.  A biologically leak-tight enclosure

2.  Sterilant and purge as transfer apparatus

3.   Monitoring equipment

4.   Access and tools for assembling

Components to be assembled and the necessary tools first are sterilized internally.   These articles then are placed in an enclosure, which is flooded with a sterilizing gas, usually a mixture of ethylene oxide and Freon 12.   Gloves resistant to the sterilizing gas are used for access to the space or, in larger facilities, assembly technicians can enter the enclosure wearing sealed clothing with outside air source and exhaust capability.   If the assembly does not include encapsulation, the assembled part must be sealed in a biologically leakproof bag before the enclosure is purged and opened.   To reduce the chance of leakage, the enclosure generally is kept at a slightly positive pressure with respect to the outside atmosphere.   There seems to be no practical limit to the size of the enclosure.

e.   Compatibility of Decelerator Materials with Spacecraft Sterilization

More materials and components suited to deployable inflatable decelerators for an entry capsule are compatible with sterilization requirements than expected.   Some ancillary components are made inherently sterile internally during manufacturing or can be made sterile by simple changes in manufacturing.   Most electronic parts fit this category.

Under JPL Contract BE4-229753, Cook Electric Co. of Chicago, Illinois, tested samples of candidate materials for deployable inflatable decelerator devices under conditions simulating sterilization test environments.

Silk was eliminated immediately and nylon was so seriously degraded by thermal sterilization that further testing was not considered.   If the thermal sterilization could be mitigated, nylon probably would withstand the chemical sterilization without serious degradation. E. I. du Pont de Nemours Company reports nylon has been subjected to ethylene oxide and to Freon 12, separately, at temperatures higher

than 104 F without serious degradation.  Therefore, the combination probably would not be critically damaging.  Nylon, after thermal cycling, was markedly stiffer and less flexible.  In contrast to preliminary tests where adhesion of unscoured materials was observed, no adhesion of the nylon to itself or to the stainless steel plates was observed.  No adhesion was observed with either dacron or Nomex.[a] Thus, long-time packed storage at elevated temperature with or without vacuum should not be a serious concern for structures made of nylon, dacron, or Nomex.  Dacron is a promising candidate for a sterilizable retardation system.  Average strength losses of all dacron configurations from sterilization and vacuum exposures did not exceed 20 percent.  Nomex is also a promising candidate material for the Mars entry retardation system.  Average strength losses of all Nomex configurations from sterilization and vacuum exposures did not exceed five percent.

Depending on the specific material and the temperature-vacuum envelope, some stiffening and heat setting of folds and wrinkles could occur with possible adverse effects on deployment.  Effects from folding and compacting were negligible.

f.  Effects of Temperature and Sustained Loadings

(1)  General Characteristics

Figures 36 and 37 present typical strength relationships for nylon, dacron, and Nomex under various conditions.  These curves show optimum strength-to-weight characteristics for undegraded materials at room temperature are obtained with nylon, dacron, and Nomex, respectively.  These yarns appear in the same order when listed according to sensitivity to initial elevated temperature values and sustained loads at room temperature (see Figure 37).  Thus, after exposure to a preload temperature of

_____

[a]A high temperature polyamide produced by I. E. duPont deNemours and Co., Inc., Wilmington, Del.
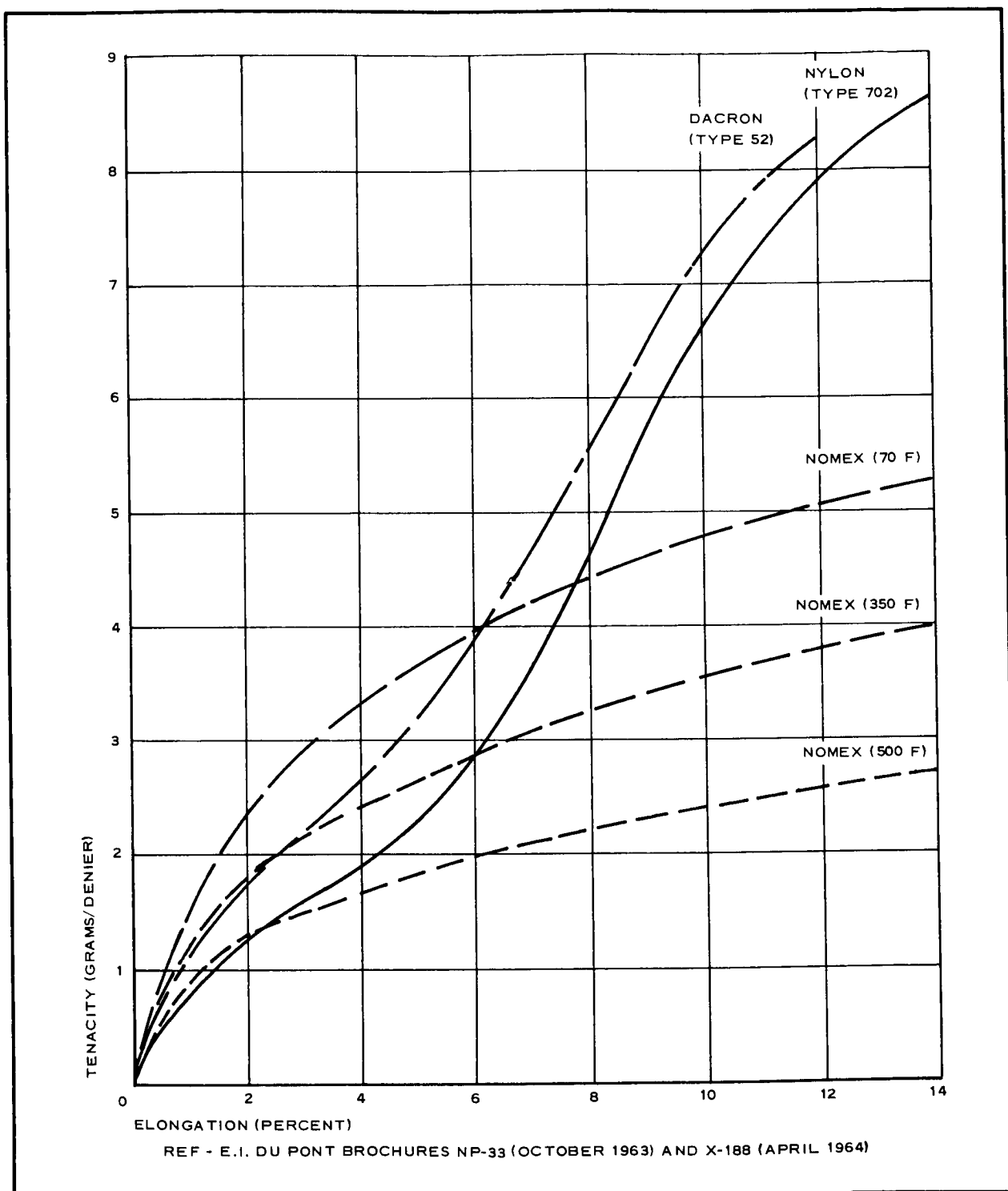
Figure 36 - Typical Stress-Strain Curves for Dacron, Nylon, and Nomex
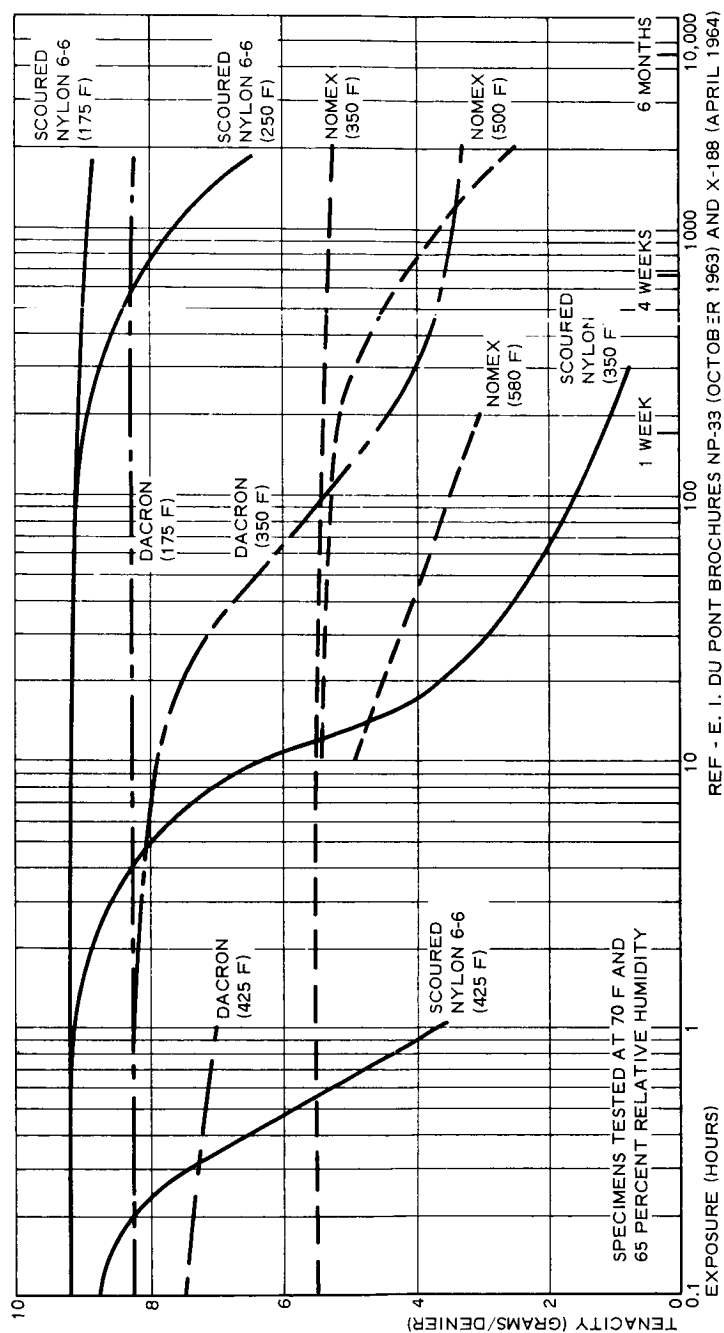
Figure 37 - Strength Retained by Nylon, Dacron, and Nomex Yarns
after Exposure to Hot, Dry Air
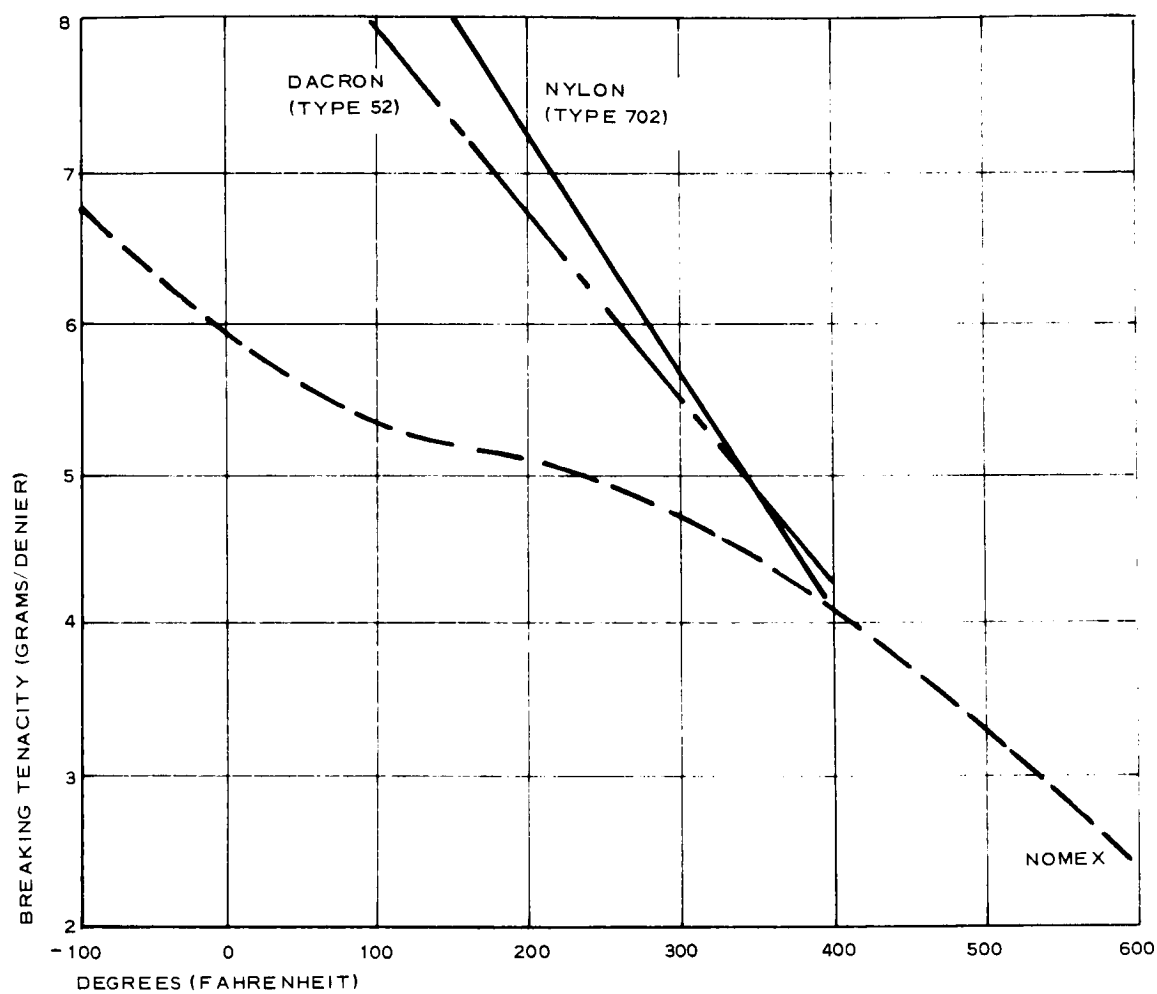
Figure 38 - Breaking Tenacity at Various Temperatures

350 F, for example, dacron is superior to scoured nylon after a loading time of 5 hr and Nomex is superior to dacron after a loading time of 100 hr. These effects become more pronounced with exposure to higher preload temperatures, with dacron having almost twice the tenacity of scoured nylon after exposure to 425 F and a 1-hr loading. Nomex remains usable after exposure to preload temperatures up to 580 F and a continuous loading time of 200 hr.

Figure 38 indicates relative performance of nylon, dacron, and Nomex when subjected to a sustained load at an elevated temperature. At a temperature of about 400 F, nylon, dacron, and Nomex, respectively, exhibit the best initial properties. Above 400 F, Nomex is the only one of the three that will withstand a sustained load.

Somewhat less temperature sensitivity was observed in unscoured nylon compared with scoured nylon. This lessening apparently resulted from the protection of the manufacturing oils. Investigations showed no difference between scoured and unscoured dacron.

(2) Stiffness

At room temperatures, nylon is more flexible than dacron and both are more flexible than Nomex for elongations below approximately six percent (see Figure 36). When stressed to breaking, however, Nomex is the most flexible, with dacron and nylon approximately 50 percent stiffer.

(3) Shrinkage

Results of heat shrinkage of virgin fiber are shown in Figure 39. Dacron reaches about 25 percent shrinkage at about 440 F; nylon reaches 16 percent at the same temperature and Nomex only 4-1/2 percent at 600 F. These shrinkages occur in about 30 sec
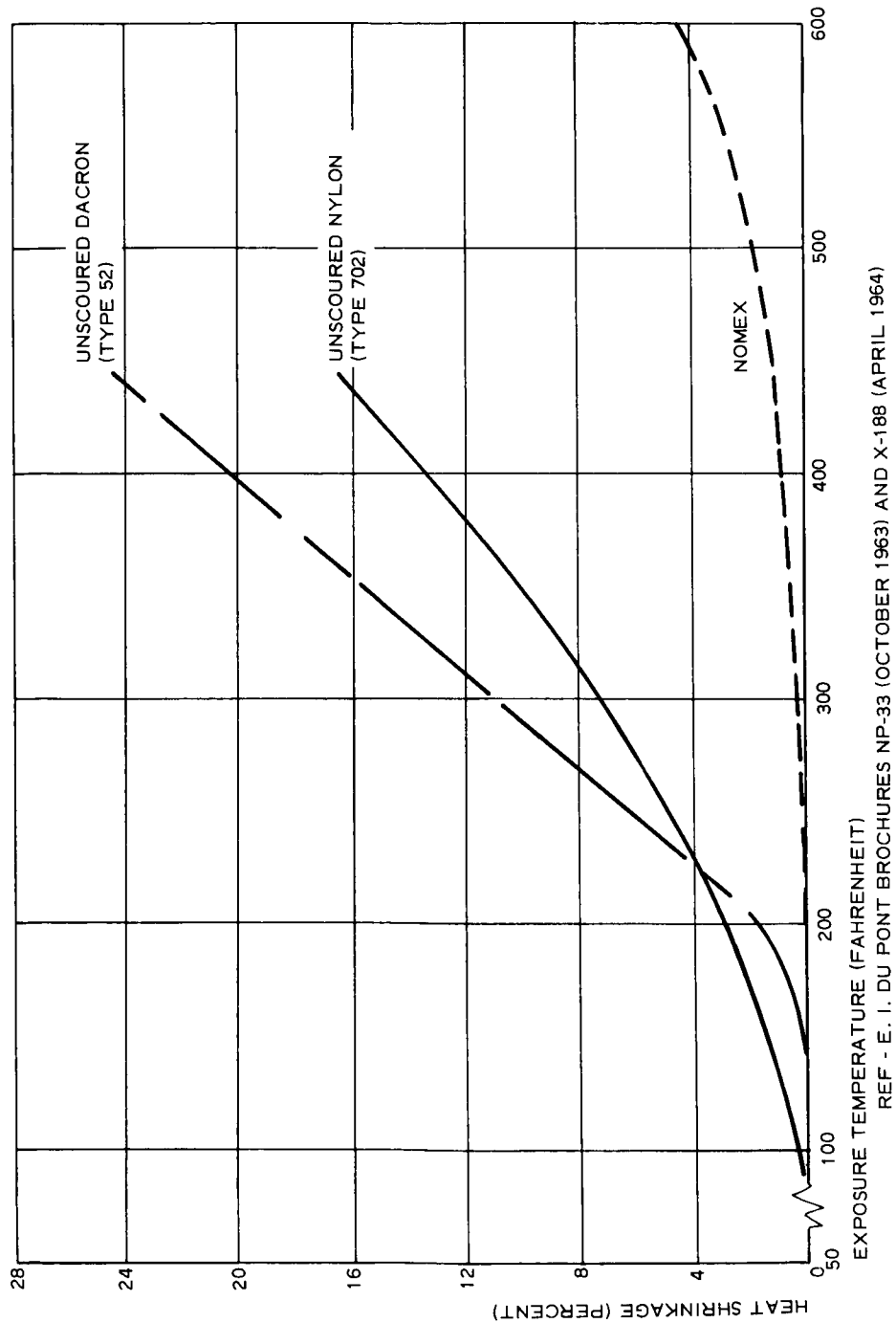
Figure 39 - Shrinkage of Nylon, Dacron, and Nomex
Exposed to Hot, Dry Air

from the time the yarn reaches temperature. While nylon will start shrinking from temperatures above 75 F, dacron does not start until temperatures above 140 F are obtained. The cross-over is at 4 percent shrinkage at 225 F. A heat-setting process must be specified in procurement of nylon and dacron before patterns are cut.

(4)  Radiation Resistance

Nylon, dacron, and Nomex are degraded in tensile strength and elongation by irradiation with ultraviolet. Their responses, maximum and minimum, are not necessarily to the same wavelengths (see Figures 40 through 42). Nylon is least affected by 369 u and most degraded by 244 $\mu$; dacron least by 369 $\mu$ and most by 314 u; Nomex least by 314 $\mu$ and most by 369 $\mu$. Nomex has the best long-range resistances. Bar graphs (see Figure 43) show how ultraviolet radiation and elevated temperatures affect Nomex.

Gamma radiation results are presented for Nomex in Figure 44. Radiation resistance of Nomex, dacron, and nylon 6-6 is presented in Table III with respect to various radiation sources.

2.  DECELERATOR MATERIALS AND COATINGS

    a.  General

    Consideration of candidate materials for expandable terminal decelerators for Mars atmospheric entry must include woven fabrics of synthetic fibers and various coating materials for porosity reduction and heat protection of woven fabrics. Low predeployment volume, low weight, and high strength are required of the decelerator. These indicate flexible structures of woven fabrics will produce the best first-stage decelerator for the entry capsule.
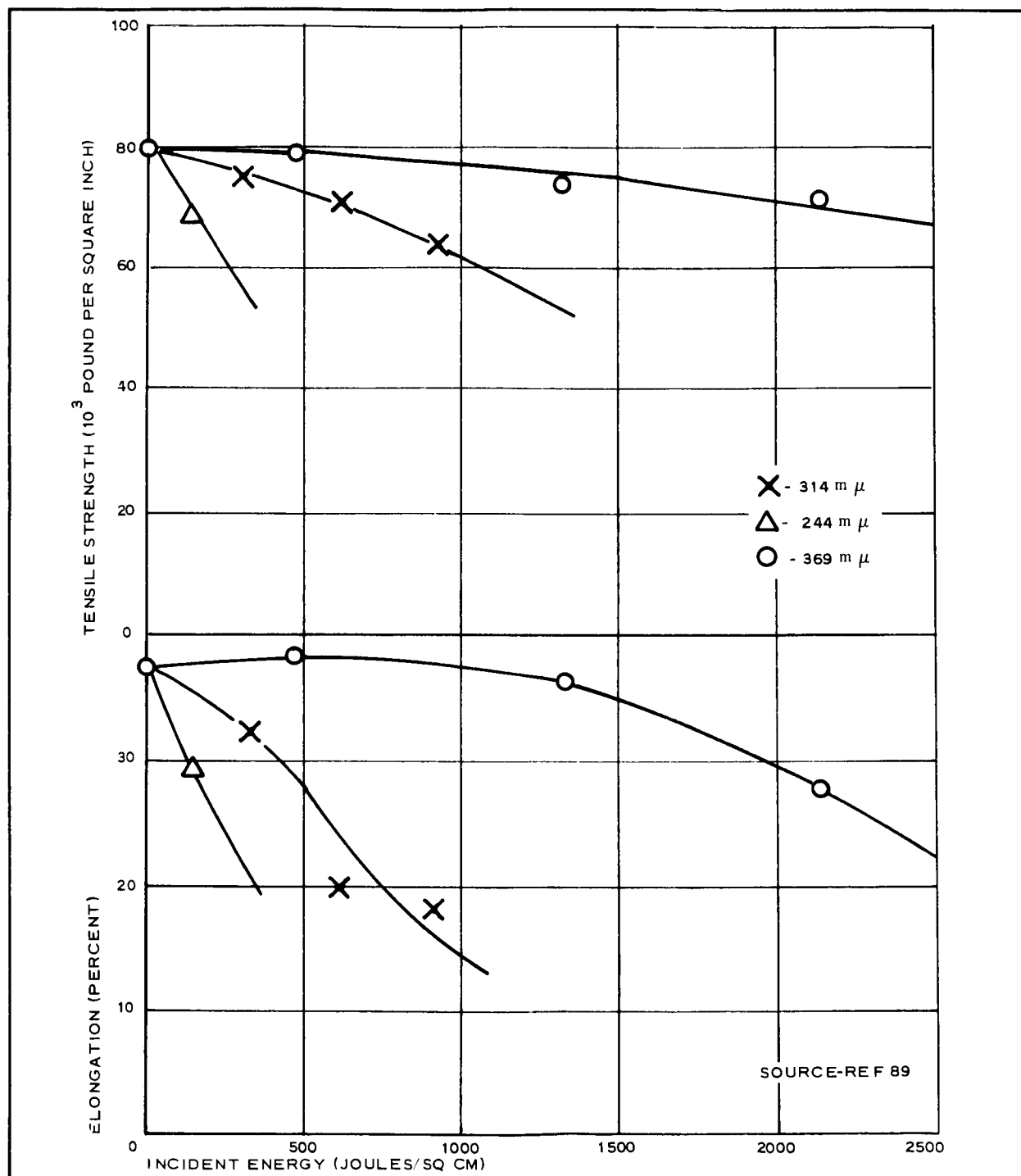
Figure 40 - Variation of Tensile Strength and Elongation of Nylon Fibers
Irradiated in Nitrogen at 244 $\mu$ , 314 $\mu$ , and 369 $\mu$
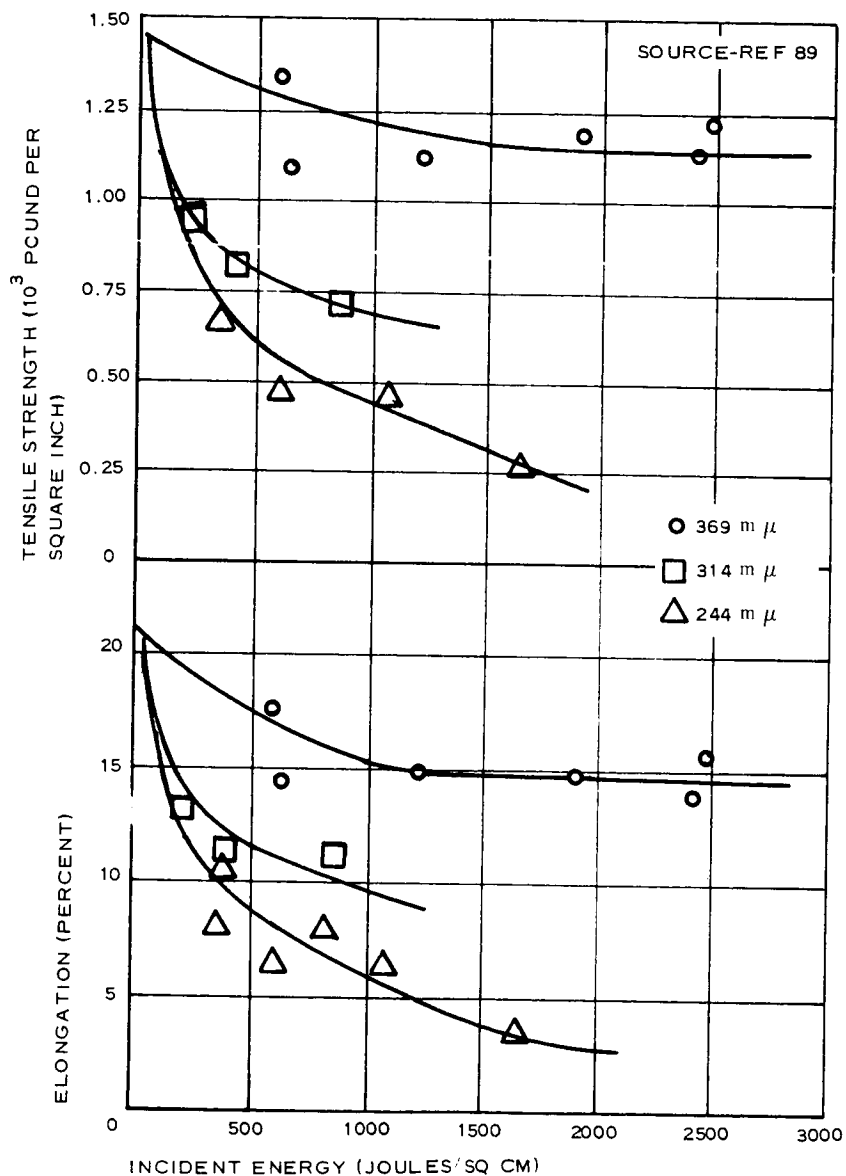
Figure 41 - Variation of Tensile Properties with Incident Energy for
Dacron Irradiated in Nitrogen with A-H6 Lamp
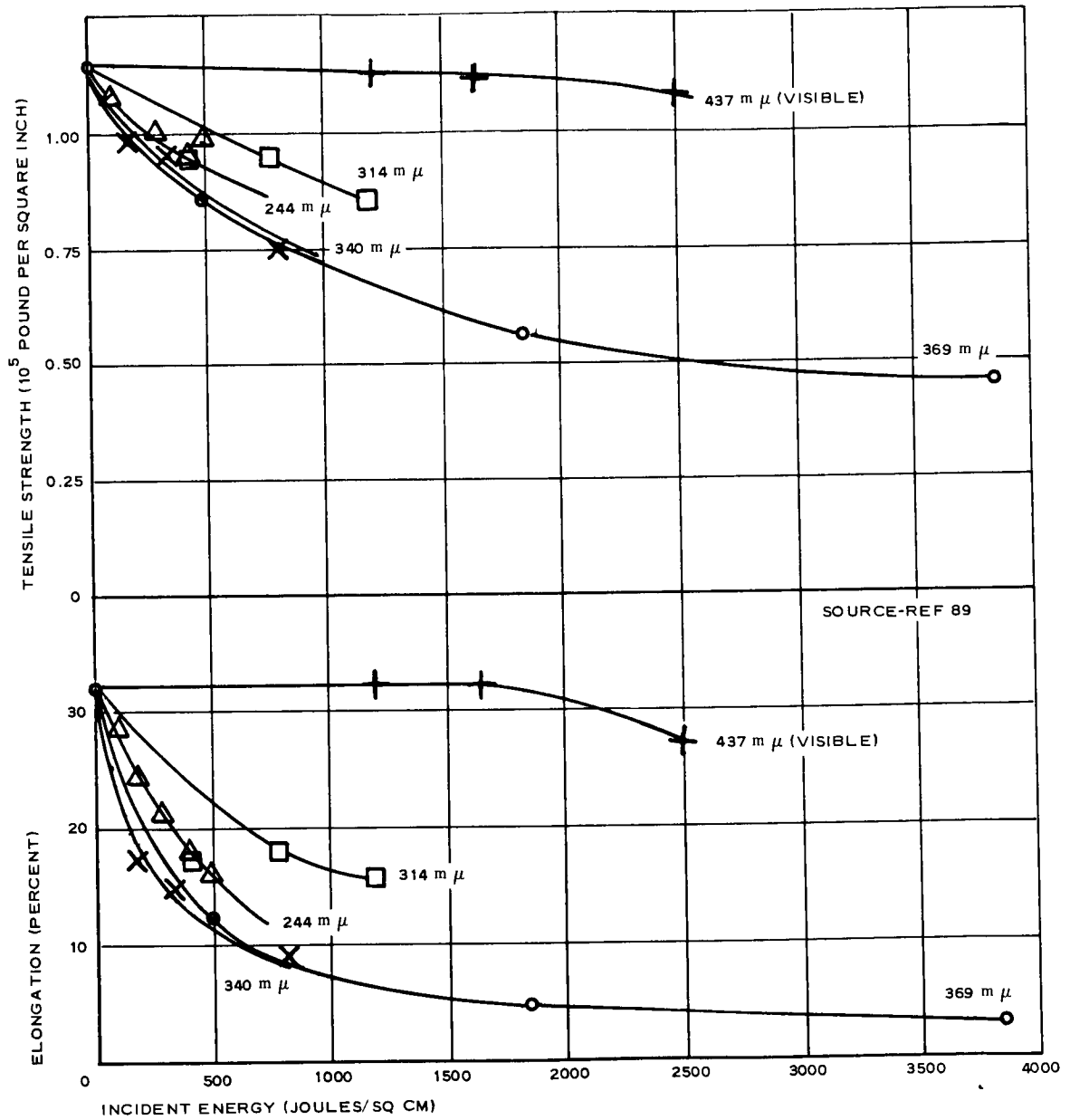
Figure 42 - Variation of Tensile Strength and Ultimate Elongation of
Nomex Irradiated in Nitrogen with Ultraviolet Light and
with Visible Light (437 $\mu$ )
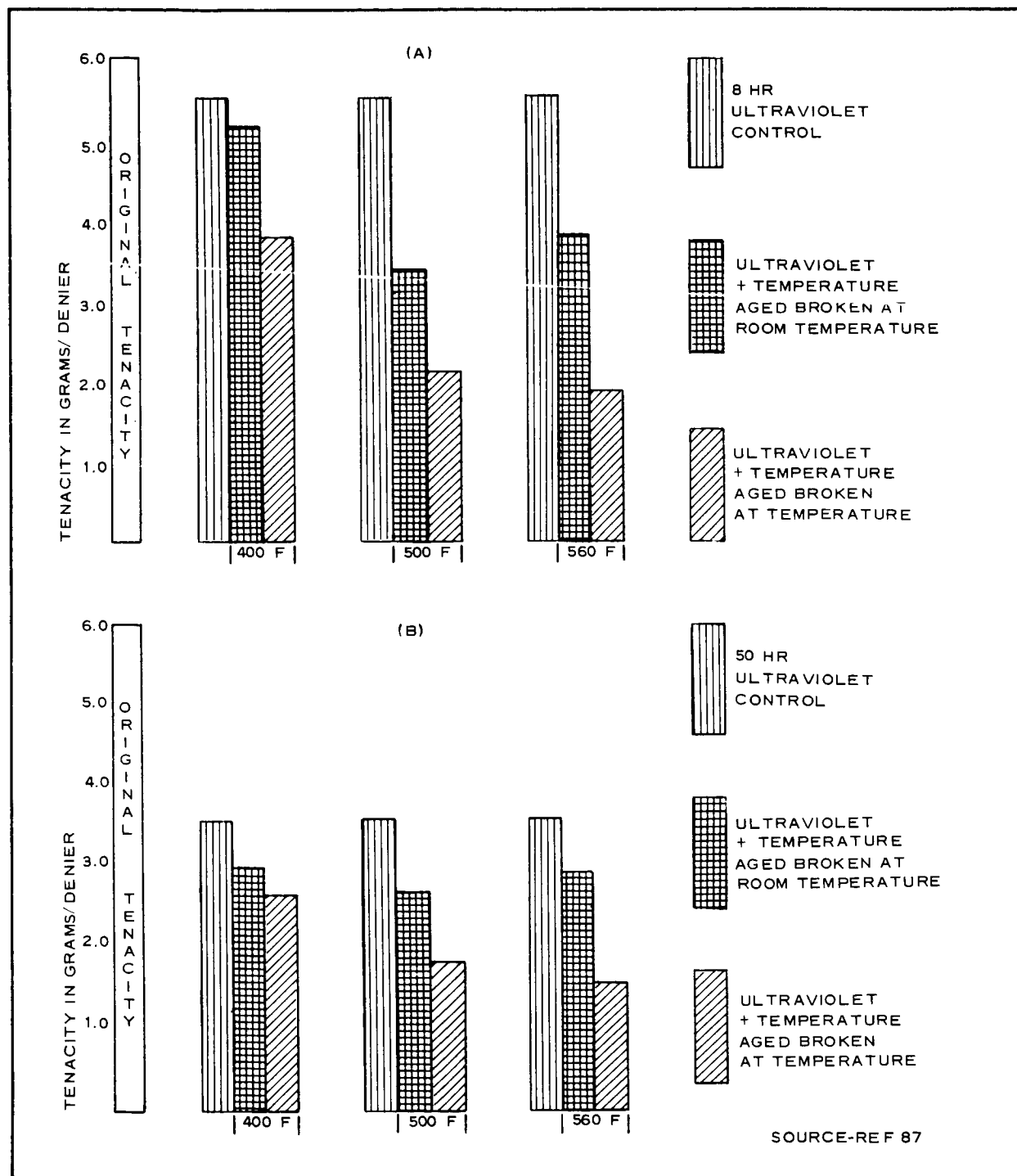
Figure 43 - Degradation of Nomex after Exposure to Ultraviolet
Radiation and Elevated Temperatures
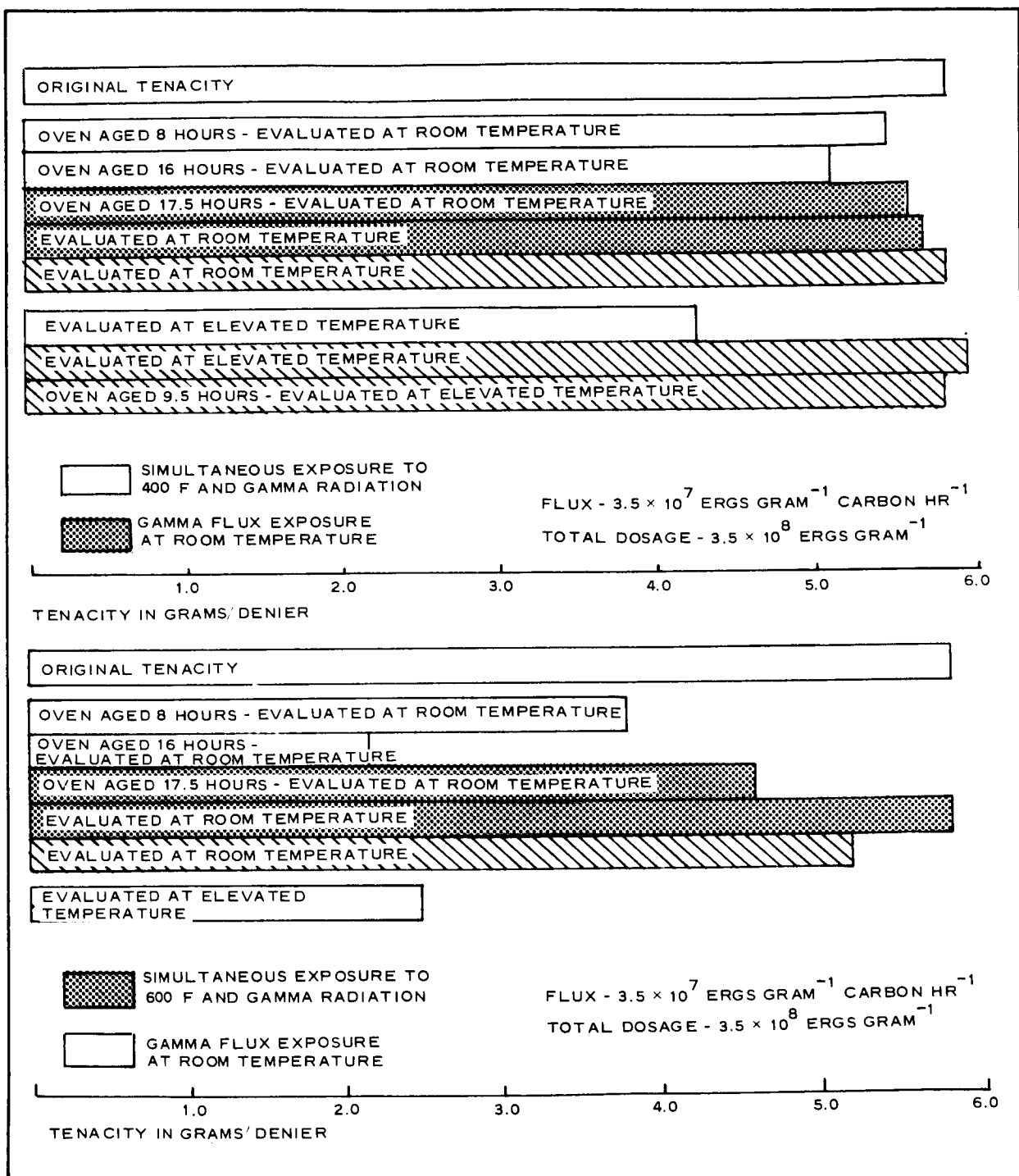
Figure 44 - Effects of Gamma Radiation on Nomex at 400 F (Top) and
at 600 F (Bottom)

TABLE III - RADIATION RESISTANCE: EFFECT OF

EXPOSURE ON YARN STRENGTH[6]

| Dosage | Tenacity retained (percent) | | |
|---|---|---|---|
| | Nomex | Dacron | 66 Nylon |
| β - Van de Graff | | | |
| 200 mega reps | 81 | 57 | 29 |
| 600 mega reps | 76 | 29 | 0 |
| X-rays (50 kv) | | | |
| 50 hr | 85 | 22 | . . . |
| 100 hr | 73 | 0 | . . . |
| 250 hr | 49 | 0 | . . . |
| Brookhaven pile (50 C) | | | |
| 200 mega reps | 70 | 45 | 32 |
| 1000 mega reps | 55 | Radioactive | Crumbled |
| 2000 mega reps | 45 | Radioactive | Crumbled |

b.   Fiber, Yarn, and Filament Material

Characteristics of woven fabrics are determined largely by the basic
fiber or filament materials. Important fiber qualities for deployable
decelerator materials include high strength, temperature resistance,
high modulus of elasticity, flexibility, abrasion resistance, and
chemical stability.

These basic properties when related to deployable, inflatable decel-
erators show that high modulus of elasticity and flexibility are con-
tradictory and are difficult problems for textile technology and pro-
duction. Available fibers with good temperature-resistance qualities
also tend to have high modulus characteristics, making temperature
resistance and flexibility difficult to attain in a single fabric.

Table IV summarizes effects of various environmental conditions on

man-made and natural textiles used or considered for aero-dynamic decelerators.

c.   Woven Fabric Materials

  (1)  General

Available woven fabric materials applicable for deployable decelerator construction, in order of increasing strength and temperature resistance, are nylon, dacron, and Nomex.

Most current applications are filled by woven fabrics of nylon, dacron, or Nomex with or without coatings. Plastic film is suitable only for low-load, low-temperature applications. Although fiberglass has high basic strength, modulus, and temperature resistance, it is notoriously vulnerable to folding damage and interfilament abrasion. Fiberglass has been used successfully in rigid applications but has been disappointing in flexible-fabric applications.

  (2)  Operational Temperature Ranges

Decelerator materials are available for operation from 300 to 1500 F (see Figure 45) Substrate material recommended from 300 to 600 F is Nomex coupled with silicon or fluoroelastomers. Between 1000 to 1500 F, substrates of stainless steel or René 41 woven cloth coated with a high-temperature coating, such as CS-105 (Goodyear Aerospace), can be used.

In other operational temperature ranges, proved materials are now unavailable. An obvious area for development of suitable materials lies between 600 to 1000 F. At temperatures above 600 F, the strength of Nomex drops rapidly; below 1000 F, the penalty in strength-to-weight ratio paid in the use of stainless steel cloth is too severe.

A potential material to fill this void is fiberglass (see Figure 45).

TABLE IV - EFFECTS OF ENVIRONMENT ON NATU

| Environment | Cotton | Silk | Viscose Rayon | F |
|---|---|---|---|---|
| Heat | Highly resistant to dry heat; yellows at 248 F; decomposes at 302 F; burns readily | Begins to decompose at 270 F; rapid disintegration above 300 F; burns readily | Loses strength above 300 F; decomposes at 350 F to 400 F; burns readily | Scorche about 20 cotton, cotton, |
| Age | Little or none | Slight yellowing and loss of tensile strength | Slight | Little o |
| Sunlight | Loses strength; formation of oxycellulose; tendency to yellowing | Loses tensile strength; affected more than cotton | Loses tensile strength after prolonged exposure; very little discoloration | Loses s to color |
| Chemicals | Disintegrated by hot dilute acids or cold conc. acids. Shells (mercerization) in caustics damaged by prolonged exposure in presence of air. Bleached by hypochlorites and peroxides, oxidized into oxycellulose by strong oxidizing agents. | Fairly resistant to weak acids; dissolved by strong acids except nitric. Insensitive to dilute alkali unless hot; dissolves in strong alkalis. Above pH11 and below pH3 stability decreases rapidly | Strong alkali causes swelling and reduces strength. Attached by strong oxidizing agents; not damaged by hypochlorite or peroxide bleaches | Disinteg lute or acids. shrinks ing. Re bleache dyehous |
| Organic solvents | Resistant | Resistant | Generally insoluble; soluble in cuprammonium | Unaffect |
| Moths | Not attacked | Attacked slightly | Not attacked | Not atta |
| Mildew | Poor resistance unless | Attacked | Attacked | Same as |

| Environment | Dacron | Glass Fiber | Polyethylene | N |
|---|---|---|---|---|
| Heat | Highly resistant to degradation and discoloration; melts at 480 F | Will not burn; strength loss starts at 600 F, continues to limiting temp. of 1000-1500 F, softens 1500 F | 5% shrinkage at 165 F; softens at 225 to 235 F; melts at 230 to 250 F; slow burning | Strength 200 F, u melts 48 |
| Age | Virtually none | None | Virtually none under normal conditions | Virtuall |
| Sunlight | Loses strength on prolonged exposure; no discoloration | None | Prolonged exposure decreases tensile strength | Moderat |
| Chemicals | Good resistance to mineral acids except conc. sulfuric. Good resistance to weak alkali, moderate to strong alkali at room temp.; dissolves in hot strong alkali. General good resistance to other chemicals; excellent to bleaches and oxidizing agents. | Good resistance to all but hot strong acids. Attacked by hot solutions of weak alkalis and cold solutions of strong alkalis. Generally good resistance. | Very resistant to acids. Generally good resistance to caustics and other chemicals. | Good re dilute ac Become mineral lis. Ger to comm |
| Organic solvents | Generally insoluble; soluble in some phenolic compounds | Insoluble | Insoluble, but swells in chlorinated hydrocarbons, aromatics | Insoluble phenols |
| Moths | Not attacked | Not attacked | Not attacked | Not atta |
| Mildew | Good resistance | Wholly resistant (binder may be attacked) | Good resistance | Good re |

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## RAL AND MANMADE TEXTILES

| rtisan | Nylon | Orlon |
|---|---|---|
| s in ironing at<br>C higher than<br>otherwise like<br>viscose rayon | Yellow slightly at 300<br>F when exposed for 5 hr;<br>melts at 482 F | Sticks at 455 F; slight<br>loss in strength after 32<br>days in air at 275 F<br>melts at 480 F |
| none | Virtually none | Virtually none |
| trength; tends | Loses strength on pro-<br>longed exposure; no dis-<br>coloration; bright yarn<br>more resistant than<br>semi-dull | Very resistant to de-<br>gradation by ultraviolet<br>light and atmosphere |
| rates in hot di-<br>old concentrated<br>Strong caustic<br>as in merceriz-<br>sistant to<br>, phenols, and<br>e reagents | Boiling in 5% HCl ulti-<br>mately causes disinte-<br>gration; dissolves in<br>cold conc. sulfuric or<br>nitric acids. Substan-<br>tially inert to alkali.<br>Generally good resis-<br>tance to other chemicals | Good to excellent re-<br>sistance to mineral<br>acids. Fair to good re-<br>sistance to weak alkalis.<br>Not harmed by oils,<br>greases, neutral salts<br>and some acid salts |
| ed | Insoluble except in some<br>phenolic compounds and<br>conc. formic acid | Unaffected by common<br>solvents |
| ked | Not attacked | Not attacked |
| for cotton | Good resistance | Good resistance (coat-<br>ing may be attacked) |

| Mylar | Nomex HT-1 | |
|---|---|---|
| reduced above<br>seful to 300 F,<br>2 F | Degrades above 700 F<br>to friable char at 482 F;<br>has 60 % room temp.<br>structural strength | |
| none | Virtually none | |
| resistance | Loses strength on' pro-<br>longed exposure; surface<br>turns bronze | |
| istance to cold<br>ids and alkalis.<br>brittle in hot<br>acids and alka-<br>erally resistant<br>on chemicals | Acid resistance better<br>than nylon 6-6; not as<br>good as Dacron or Orlon;<br>degraded by strong alkali<br>at elevated temperature | |
| ; degraded by<br>nd cresols | Highly resistant to most<br>hydrocarbons | |
| ked | Not attacked | |
| istance | Good resistance | |

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Figure 45 - Strength Retention versus Temperature for
Present and Future Fibers

However, because of its self-destructive nature under flexing of pulsating loads, fiberglass does not lend itself to this application. Efforts have been underway for some time to minimize the abrasive nature of fiberglass, with moderate success. One approach to this problem has been to impregnate yarns with elastomers to prevent adjacent filaments in the yarn from rubbing. Another approach has been the development of Beta glass fiber.[a] The Beta fiber is an extremely fine filament. Both methods have improved performance of fiberglass in flexible applications; however, this work must be continued to realize its potential.

Another possibility of filling the substrate-material void from 600 to 1000 F is to extend the capability of Nomex. The most practical way to do this is to protect the substrate so its temperature does not exceed 600 F. Coating the substrate with ablative offers this protection, but the thickness required and the associated rigidity of thermal insulators as they are now known make this use prohibitive. Development of low-temperature ablators has caused a considerable lag in development of high-temperature ablators. To obtain ablative materials that will satisfy the requirements for use on decelerators is a definite problem area that warrants investigation.

d. Fabric Coatings

Coatings are applied to reduce fabric porosity or to protect the basic fabric from high aerodynamic temperatures or other heat load, or both. No rigid line of demarcation exists between coatings for gas tightness and those for heat protection. Urethane and silicon rubbers have higher temperature resistance than neoprenes; however, any coating with a low thermal conductivity can render a limited heat protection. The formation of an insulating char is sometimes a usable mechanism. For transients, an aluminum metalizing can be adequate,

[a]A product of Owens-Corning Fiberglass Corp., Toledo, Ohio

but when temperature requirements are high and fairly long, coating performance can become quite sophisticated. Heat protection can be afforded by combining mechanical and chemical reactions to the temperature through char formation, sublimation, and ablation.

Coating materials are available that perform satisfactorily to 1500 F. To 600 F, fluoroelastomers perform quite well for short periods of time. To 1000 F, silicones are available. Between 1000 and 1500 F, CS-105 high-temperature flexible coating is recommended. This coating, a silicon binder with a glass frit filler, has been used to 1500 F.

CS-105 coating resembles silicon elastomer at room temperature. As the temperature is raised, a thermal decomposition of the elastomer and fusing of the glass frit occur. The weight of decomposition is a time-temperature phenomenon that progresses slowly at 800 F and increases as the temperature rises. The glass frit does not fuse until 1100 to 1200 F has been reached. Hence, between approximately 800 to 1100 F the elastomer is decomposing and the glass frit does not fuse . During this transition phase, the coating is most susceptible to damage and the gas permeability increases. When the coating is subjected to a heat flux of such magnitude, it must traverse the critical temperature range in a relatively short time. The glass frit fuses before the silicon elastomer decomposes excessively, provides a carrier for the silicon residue, and forms an adequate gas barrier.

Elastomer coating materials applicable for reducing porosity in decelerator structure fabrics are described in Table V. Performance is evaluated for each material in a variety of environmental conditions so material limitations are readily apparent.

Some typical coatings, used or investigated for possible use in decelerators, are shown in Table VI. Neoprene, DC-131, and Viton have been used satisfactorily on flight-tested decelerators in the

low-to-medium temperature regime - that is, up to about Mach 2 and at altitudes above 70,000 ft.

The remainder of the listed coatings are potential candidates to extend the operating capability of state of the art fabrics. In many applications, ablative coating materials may be appropriate because of the short deceleration times characteristic of deployable decelerators.

The approach used to increase the temperature capability of the coating between the thermal degradation temperature of the elastomer and the desired operating temperature is to load the elastomer with a low-melting point inorganic material. As the temperature rises, a thermal decomposition of the elastomer occurs, and inorganic material changes to a very viscous fluid. The viscous fluid requires a high surface tension to hold the residue of the elastomer in suspension and to maintain a continuous film. After cooling, the material solidifies, forming a solid gas barrier with some flexibility although much less than the original unfired coating.

e.  Joining Methods

Construction of foldable, packageable structures involves the classic problem of building compound shapes from plane material. Aerospace decelerators are assembled by sewing a multitude of individually patterned pieces of a plane fabric material. These pieces are seamed together, usually by sewing with threads of material similar to the fabric filaments. BALLUTE gore patterns, for example, are cut on the bias from "square" cloth that has equal or nearly equal strength and elongation in warp and fill. Gores are alternated right and left bias to balance out differences in elongation in the warp and fill directions. So far, BALLUTEs and parachutes are of single-ply construction.

Joining methods include sewing, cementing, and, with metal fabrics, spot welding. With plastic film material, cementing is almost invariably involved, either a heat-sealing or solvent-sealing cementing or

TABLE V - RELATIVE GENERAL PROPERAL P[

| Elastomer types | Ten-sile | Tear | Abra-sion | Impact (fatigue) | Flame | Heat (F) | Cold (stiff) | Cold (brittle) (F) | Ozone | Radi-ation |
|---|---|---|---|---|---|---|---|---|---|---|
| Natural rubber | AB[+] | B | AB | AB | D | CD[+] +250 | B | B -80 | D | BC |
| Styrene butadiene rubber (Buna S or GRS) | B | BC | AB | AB | D | C +275 | BC | B -80 to -90 | D | BC |
| Isobutylene iso-prene rubber (Butyl or GR-1) | C | B | B | C | D | BC +300 | C | BC -50 to -80 | AB | D |
| Chloroprene rubber (Neoprene or GR-M) | B | B | AB | B | B | C +300 F | C | BC -45 to -70 | AB | CD |
| Polyurethane elastomers (Adi-prene, Chemigum SL, CX-1046) | A | A | A | B | CD | C +250 | C | A -30 to -95 | A | B |
| Nitrile butadiene rubber (Buna N) | BC | BC | AC | C | D | B +275 | BC | BC -80 to -90 | D | BC |
| Silicone rubbers | D | CD | CD | D | C | A +550 | A | A -200 | A | D |
| Chlorosulfonated polyethylene (Hypalon) | BC | BC | AB | BC | B | BC +325 | C | B -70 to -80 | A | BC |
| Fluorinated elas-tomers (Fluorel, Kel-F, Viton) | BC | BC | B | BD | A | A +450 | D | BC +10 to -40 | A | BC |
| Organic polysul-fide rubbers (Thiokol, GR-P) | D | D | D | D | D | C +200 to +275 | C | B -60 to -80 | A | BC |

*Source - Reference 8

[+]A = exceptional, outstanding, or excellent; B = good; C = fair; D = poor.

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ROPERTIES OF ELASTOMERS

| Gas retention | Resistance - oil, weather, chemical | Unsuitable for | Specific gravity |
|---|---|---|---|
| B | Highly resilient, low hysteresis, general | Contact with oils, ozone, strong oxidizing agents. | 0.93 |
| B | General purpose rubber, not so resilient as natural, better resistance to aging. | Contact with oil, ozone, strong oxidizing agents. | 0.94 |
| A | Weather, heat, ozone, chemical, and solvent resistant, low air permeability. | Contact with oils. | 0.92 |
| AB | Weather resistant, fair oil resistance. | Temperature extremes, contact with aromatic oils and most fuels, long exposure to low temperatures | 1.24 |
| A (R. T.) | Superior abrasion resistance, sunlight and ozone resistance, good oil resistance. | Contact with steam or hot water | 1.05 to 1.17 |
| B | Medium to good oil resistance, fair fuel resistance | Contact with ozone, strong oxidizing agents. | 0.99 |
| B | Resistant to temperature extremes, fair oil resistance, properties constant from 60 F to 500 F. | Contact with high pressure steam, aromatic oils, fuels, abrasion. | 1.25 |
| AB | Weather, heat, ozone, and moderate oil resistance, good color possibilities. | Aromatic oils and most fuels. | 1.10 |
| A | Resistant to oxidizing acids, fuels containing up to 30 percent aromatics, ozone, weahter; excellent oil resistance. | Contact with diester lubricants, uses where material must be easily flexed at temperatures below 0 F. | 1.40 to 1.85 |
| A | Excellent oil resistance, good resistance to aromatic fuels, excellent weather and ozone resistance. | Resistance to compression set particularly at temperatures above 100 F, uses where mercaptan odor would be objectionable, contact with oxidizing acids. | 1.25 to |

## TABLE VI - ELASTOMERIC COATINGS

| Coating | Vendor | Specific gravity | Specific heat | Thermal conductivity | Emissivity | Remarks |
|---|---|---|---|---|---|---|
| Neoprene 1137-C | Goodyear | 1.30 | 0.35 Btu/lb-deg F | 0.14 Btu/hr-ft-deg F | | Low temperature coating |
| DC-131 | Dow-Corning | 0.94 | 0.3-0.35 Btu/lb-deg F | 0.167 at room temperature Btu/hr-ft-deg F | | |
| Viton | Du Pont | 1.85 to 1.90 | 0.395 Btu/lb-deg F | 0.117 at 300 F Btu/hr-ft-deg F | | |
| CS-105 | Goodyear | | | | 0.92 at 1000 F | High temperature coating |
| D-65 | Dyna-Therm | 1.1 | 0.25 (68 to 150 F) Btu/lb-deg F | 0.053 at 150 F Btu/hr-ft-deg F | | Ablator |
| RTV-88 | General Electric | 1.48 | 0.35 Btu/lb-deg F | 0.18 at 200 F Btu/hr-ft-deg F | | |
| RTV-560 | General Electric | 1.42 | | 0.19 at 315 F Btu/hr-ft-deg F | | Ablator |
| AVCOAT II | AVCO | 1.0 | 0.43 (65 to 240 F) Btu/lb-deg F | 0.10 at 250 F Btu/hr-ft-deg F | | Ablator |
| DC-325 | Dow-Corning | 0.87 | 0.32 (77 to 200 F) Btu/lb-deg F | 0.08 at 200 F Btu/hr-ft-deg F | 0.90 at 70 F | Ablator |

adhesive application. Soft fabrics usually are sewed. But cementing is sometimes practical, especially with an elastomer-coated cloth, and frequently this results in higher joint efficiencies. Combinations of cementing and sewing have been used. Cementing has been used for sealing holes made by sewing or preventing the slippage of stitching. Joining of metal cloth has been by multiple staggered row spot welding. The Air Force has reported sewing such fabric sucessfully with wire thread.

The development of the optimum seam for a specific application is a matter of design and test development. Involved are such factors as the cloth and its basic strength to be developed; the coating, and an orthogonal or biased seam. Variations are possible with number of parallel rows of stitches, number of stitches per inch, type of stitch, presence or absence of reinforcing tapes or webs, types of seams, and possible overlay of elastomeric coating. The final bulkiness of the seam is a consideration with its consequent influence on packageability and deployment.

Seams invariably cause problems. Seaming represents a sizeable part of construction cost and contributes significantly to the variation in quality of the finished article. The seam adds bulk and weight, reduces flexibility, adds distortions when the structure is loaded, and almost never can be designed to develop 100 percent material strength. Although cementing generally can produce a better joint, it also is affected more adversely by elevated temperature.

f.    Material Selection and Qualification

Materials for decelerators are selected after analysis of design factors and fabrication technology. Static and dynamic loading, thermal loading, maximum size and weight of structure, and aerodynamic drag requirements dictate requirements for strength and temperature resistance. Material selected must meet these requirements after fabric strength-to-weight ratio, thickness, porosity, flexibility,

and coatings have been considered.  Material selection usually will involve fiber or filament material and size, weave, seam construction, production sequence, and other production techniques.  Different materials and fabrication techniques may be required for structure components.

# SECTION IV - DECELERATOR ENVELOPE SIZE ESTABLISHED BY

## THERMAL REQUIREMENTS

On the basis of heat-rating profiles developed in Section II, fabric weight requirements for several entry cases were estimated.

Fabric weight per unit surface area was estimated from heat-flux rate profiles on the basis of material heat capacity. This procedure, carried out in the deployment Mach range for Atmospheres VM7 and VM8, includes trajectories computed for different drag parameters using similar deployment conditions. The heat-balance procedure simply equated the cold-wall heat, presented above, and the heat capacity of the fabric material to the maximum temperature at which it retains load carrying capability. Hence,

$$q = \int \dot{q} d\tau$$

$$= \rho \delta (T_f - T_i)$$

where

$q$ = heat flux $(Btu/ft^2)$,

$\dot{q}$ = heat flux rate $(Btu/ft^2\text{-sec})$,

$c$ = specific heat of material $(Btu/lb\text{-deg F})$,

$T_i$ = initial temperature - 70 F,

$T_f$ = final maximum temperature (deg F),

$\delta$ = thickness of material (ft),

$\tau$ = time (sec), and

$\rho$ = density of material (pcf).

This heat balance assumes that thermal conductivity of the fabric material is infinitely high and that outward surface radiation is zero. Infinite thermal conductivity is a poor assumption since the usual value of this fabric thermal parameter is quite low. Surface temperature would be expected to rise quickly while temperature at the back or interior surface would change little. However, heating rates appear so low and of such short duration that this assumption is realistic for a preliminary design weight estimate. In addition, as the outer surface temperature rises, incoming heat pulse decreases because of differential temperature reduction. Furthermore, as the surface temperature rises, the back radiation can be expected to increase. Because both factors have been neglected, it will be assumed that two trends either compensate for each other or produce conservative estimates of the fabric weight required. These results were supplemented with computer solutions (see Appendix B of Vol I) using transient one-dimensional heat conduction in the fabrics to account for these parameter variations. Such solutions yield more realistic design configurations for the temperature profiles in the materials.

Results of the computations are presented in Figures 46 through 52. Fabric weight requirements in the VM7 atmosphere are increased considerably over those required for the VM8 atmosphere. This is primarily caused by increased ambient temperatures associated with this atmosphere and secondarily to the more gradual decrease of density with altitude. In addition, most decelerations in the VM8 atmosphere take place at altitudes below 50,000 ft while those in the VM7 atmosphere are between 60,000 and 110,000 ft. As a result, deceleration times in the VM7 atmosphere are longer, a fact reflected in heat input to the fabric. Requirements thus are increased significantly in the upper deployment Mach regime.
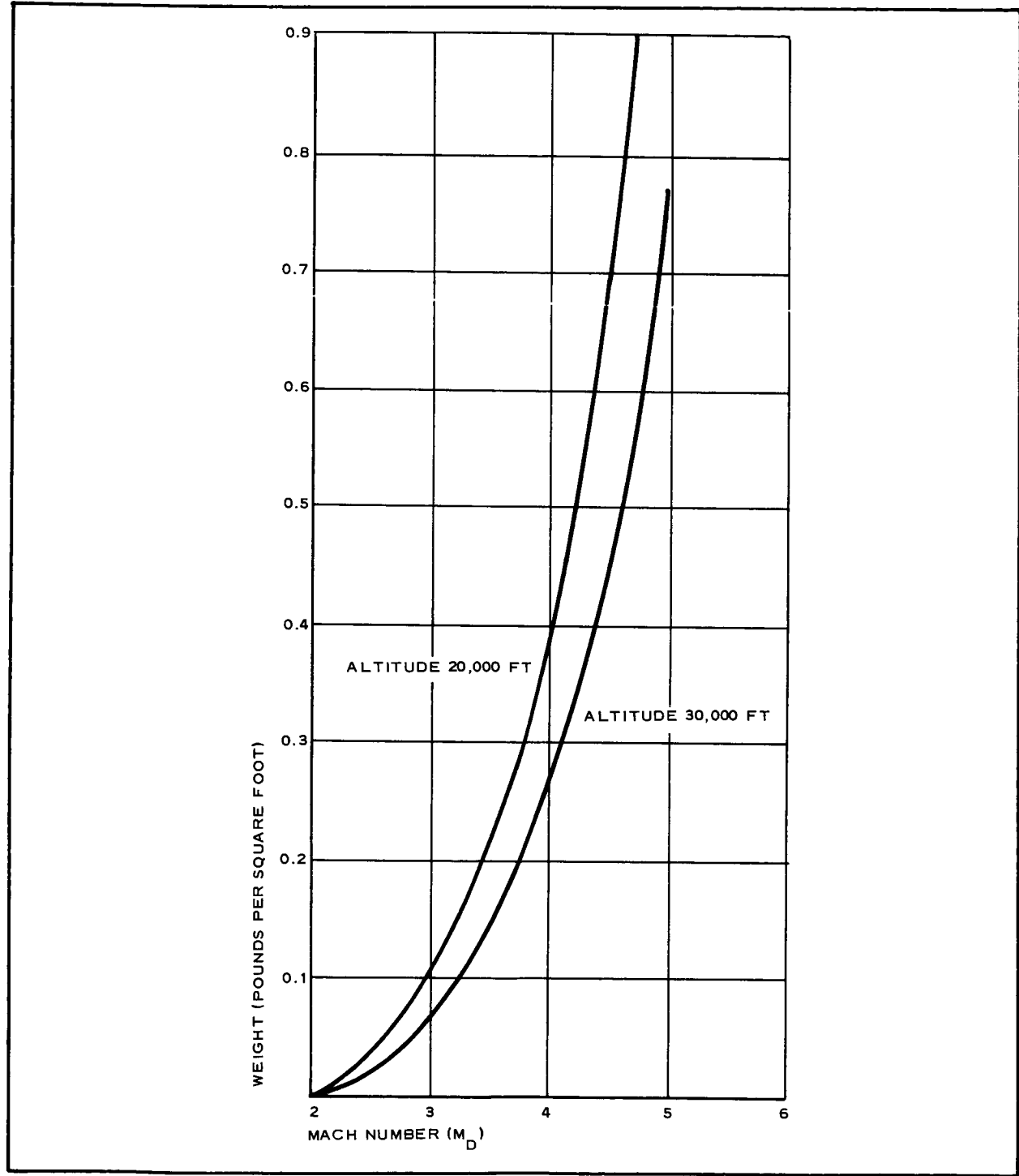
Figure 46 - Decelerator Fabric Envelope Weight Required versus
Initial Mach Number, Trajectory B1
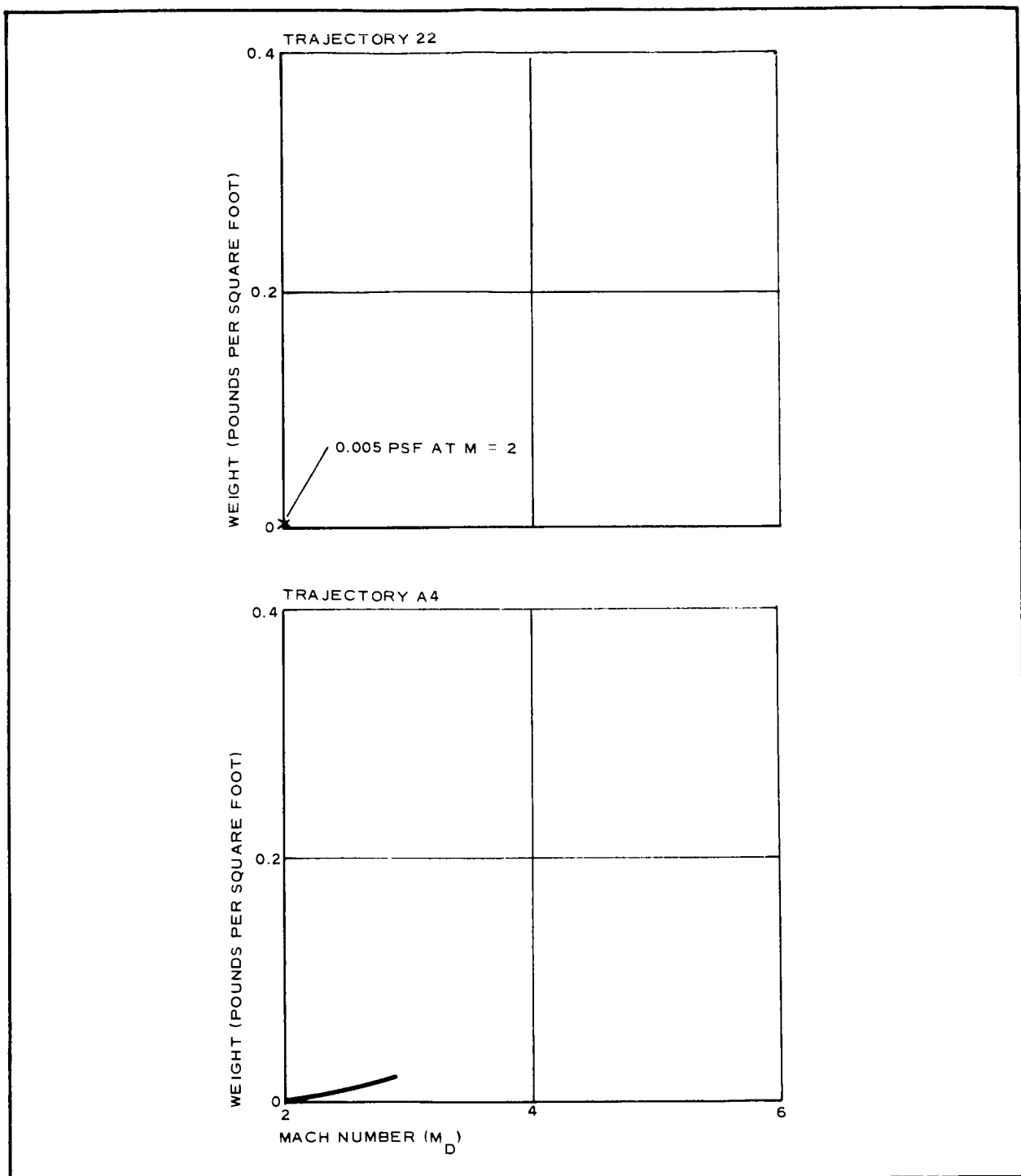
Figure 47 - Decelerator Fabric Envelope Weight Required versus
Initial Mach Number, Trajectories A4 and 22
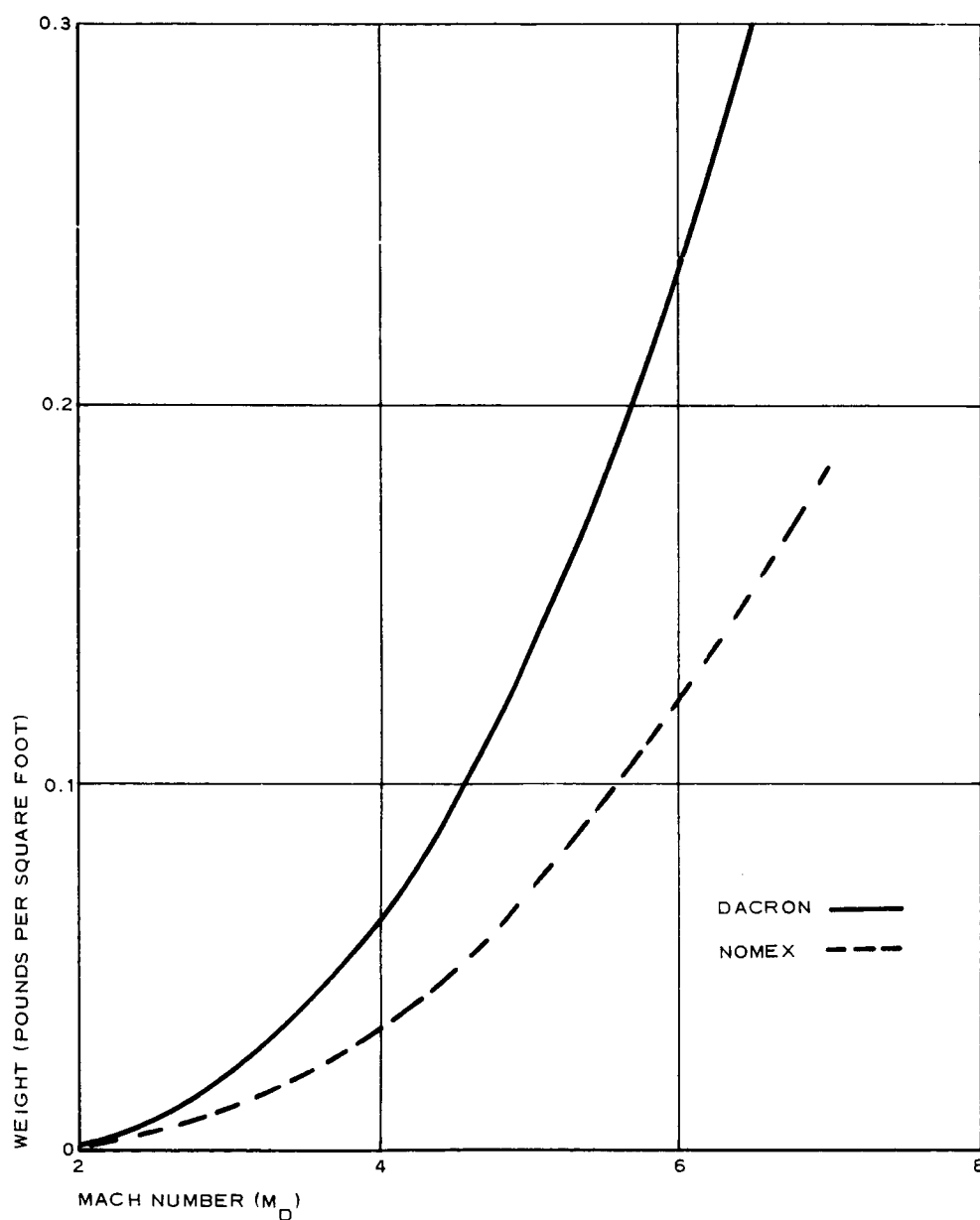
Figure 48 - Decelerator Fabric Envelope Weight Required versus
Initial Mach Number, Trajectory 37
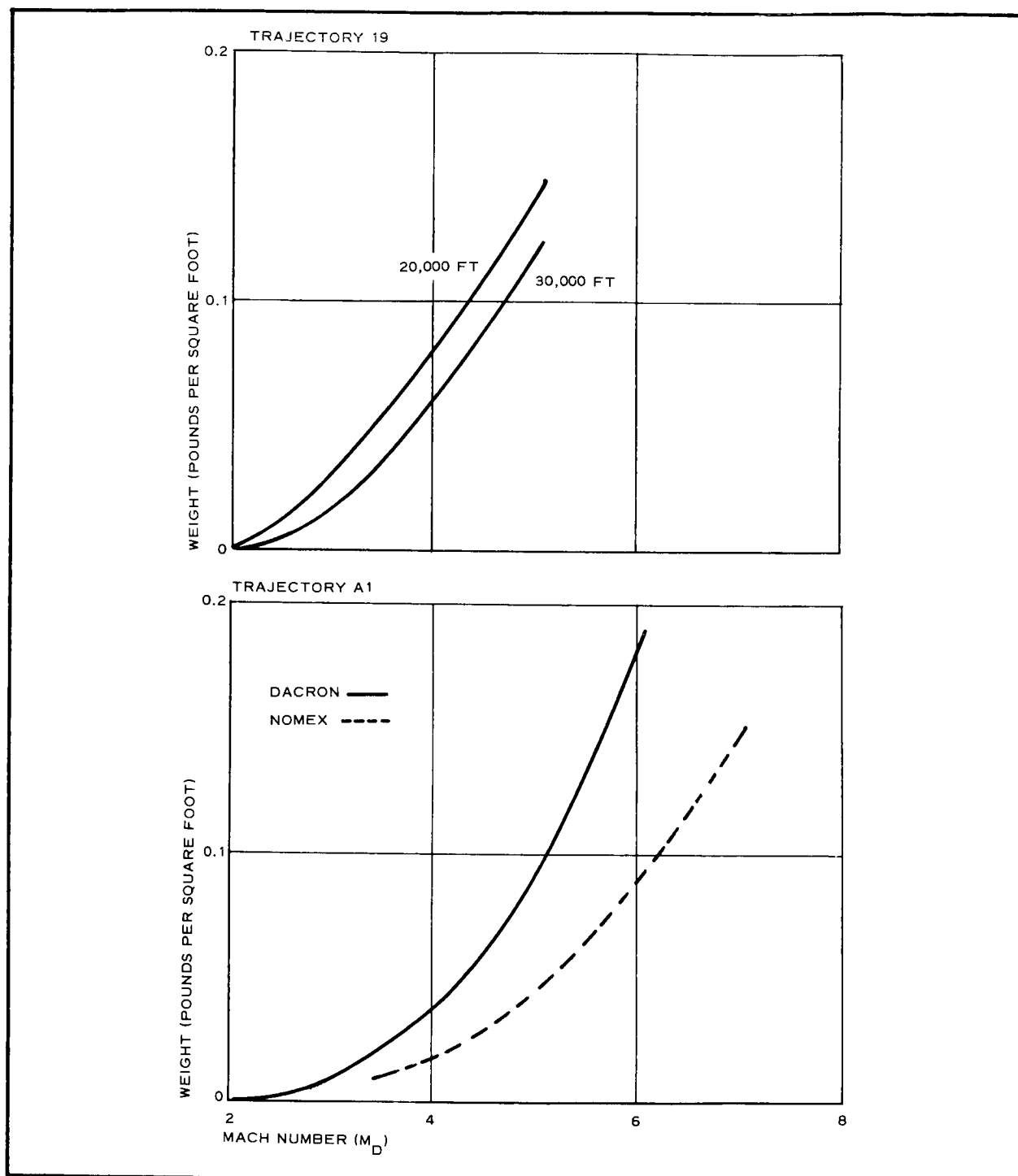
Figure 49 - Decelerator Fabric Envelope Weight Required versus
Initial Mach Number, Trajectories A1 and 19
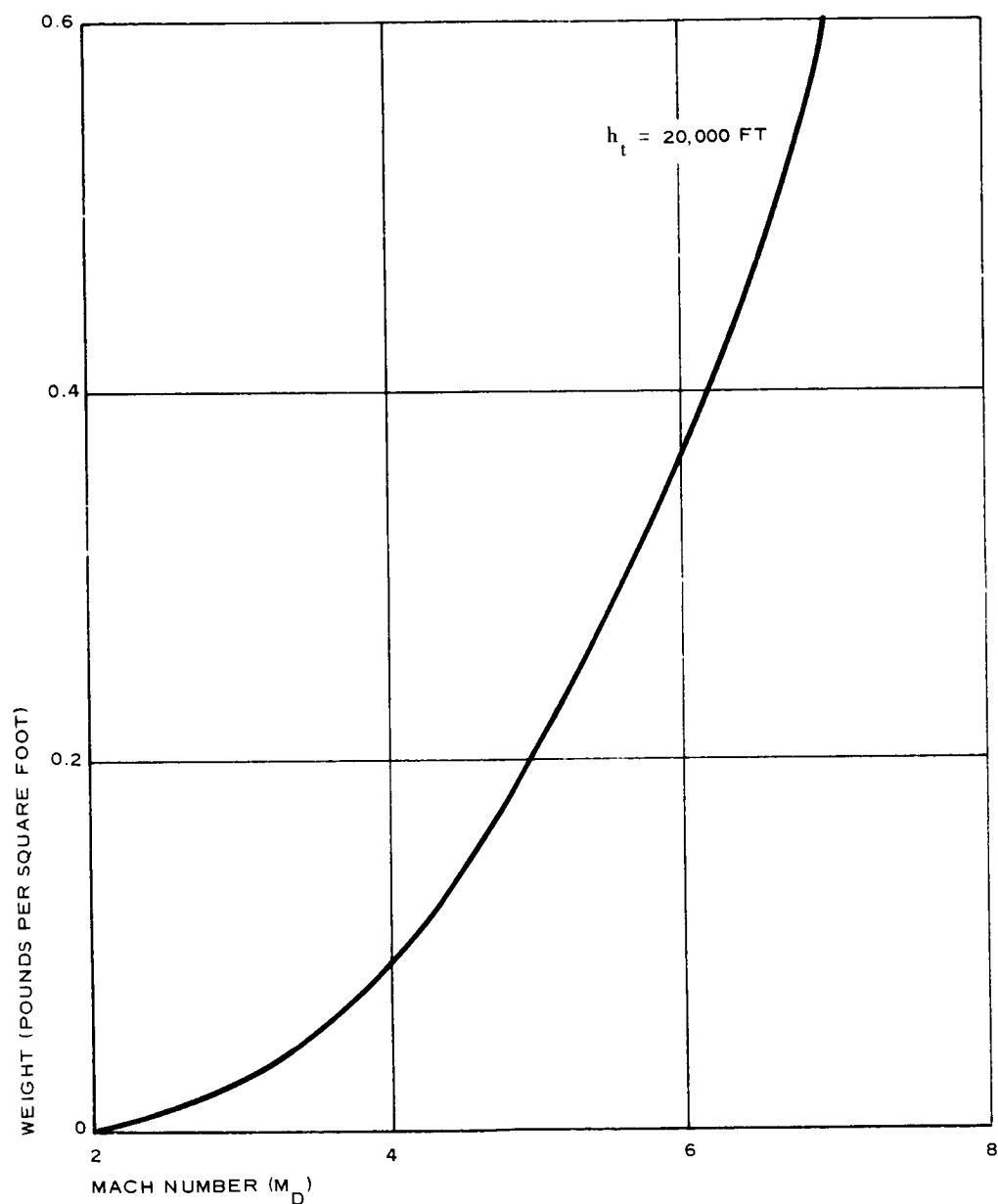
Figure 50 - Decelerator Fabric Envelope Weight Required versus
Initial Mach Number, Trajectory B3

I.

TARGET MACH NO.

BASIC VEHICLE TRAJECTORY

$n_1$
$n_2$
$n_3$
$n_1$
$n_2$
$n_3$

ALTITUDE, h

DECELERATOR INITIAL OPERATING MACH NUMBER, $(M_I)$

MACH NUMBER.
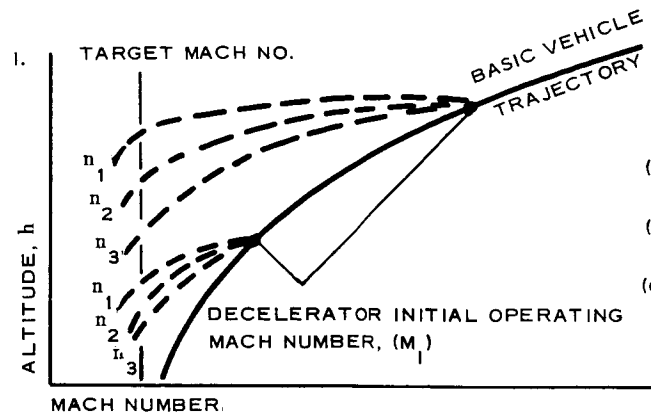
$$n = \text{PERCENT OF} \frac{(C_D A)_T}{(C_D A)_V} \times (C_D A)_V$$

$(C_D A)_V$ = VEHICLE DRAG AREA $\doteq$ 100 PERCENT VALUE

$(C_D A)_D$ = DECELERATOR DRAG AREA

$(C_D A)_T = (C_D A)_V + (C_D A)_D$

II.

TARGET ALTITUDES

TRAJECTORY 19

A1

ALTITUDE, h

PERCENT OF $(C_D A)_T / (C_D A)_V$
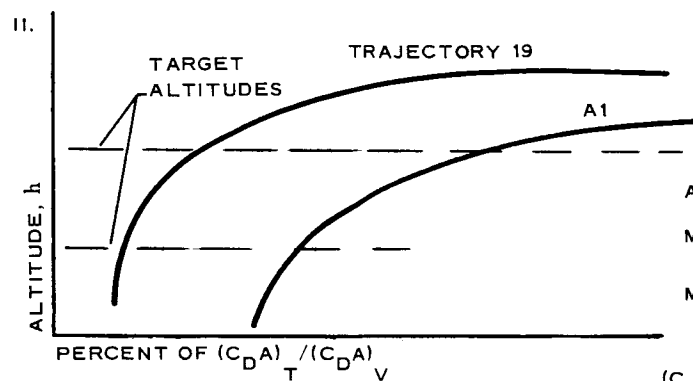
ATMOSPHERE VM8

$M_T = 1.0$

$M_I = 3.0$

III.

TARGET ALTITUDE

PERCENT OF $\dfrac{(C_D A)_T}{(C_D A)_V}$

300
275
250
225
200

ALTITUDE, h

INITIAL MACH NUMBER, $M_I$

ATMOSPHERE VM8

TRAJECTORY 19

$M_T = 1.0$

$(C_D A)_V = 281.5$ SQ FT

IV.

ATTACHED BALLUTE

$C_D$

MACH NUMBER

V.

$D_D$

$M > 4.0$
3.0
2.0

DECELERATOR DIAMETER,

ATTACHED BALLUTE

TRAJECTORY 19

$(C_D A)_V = 281.5 \text{ SQ FT}$

PERCENT OF $(C_D A)_T / (C_D A)_V$

VI.

$D_D$

DECELERATOR DIAMETER,

ATTACHED BALLUTE

ATMOSPHERE VM8

TRAJECTORY 19

$M_T = 1.0$

$h_T = 20,000 \text{ FT}$

INITIAL MACH NUMBER, $M_I$

Figure 51 - Six Steps in Determining Decelerator Size
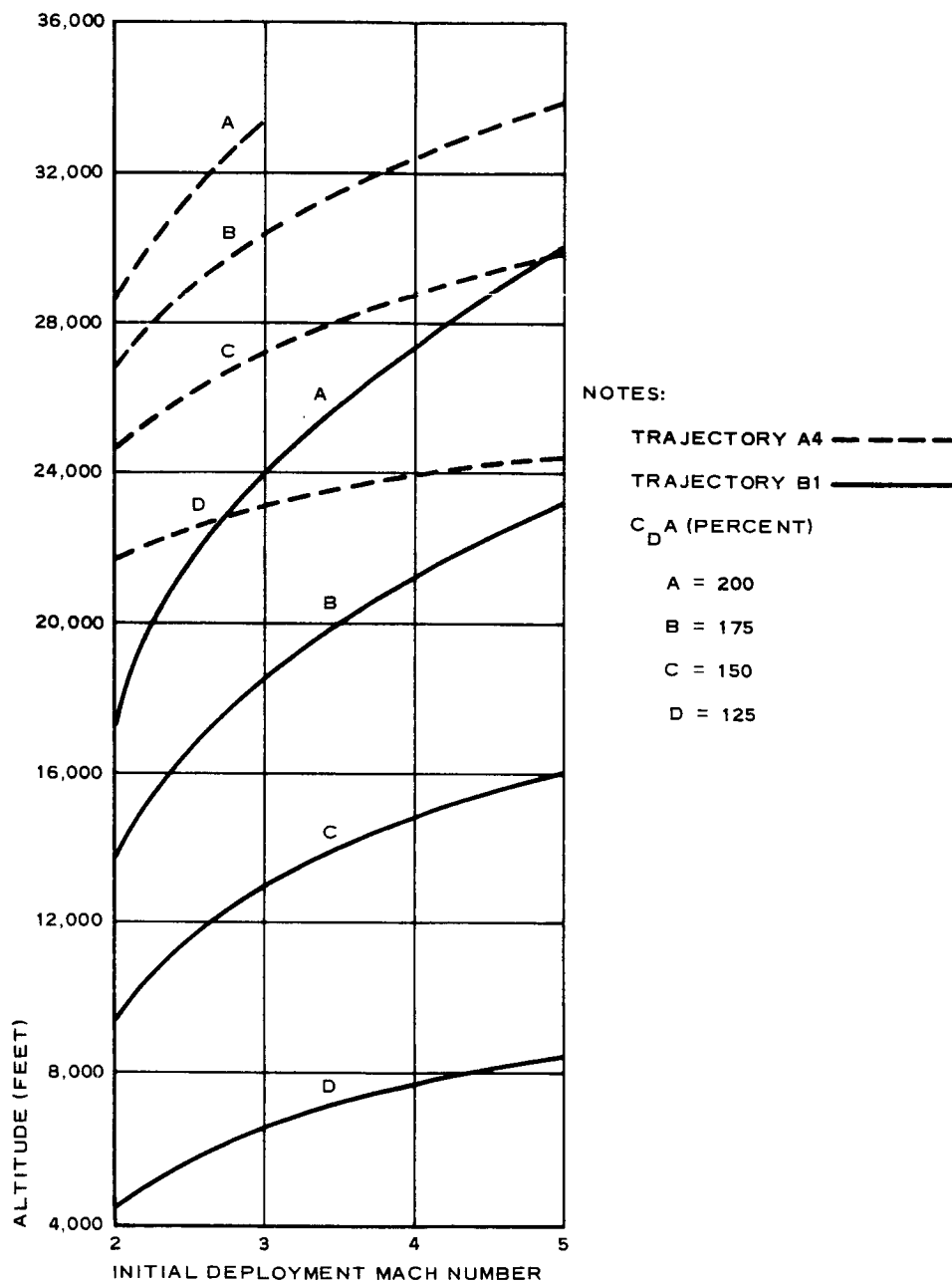
Figure 52 - Altitudes at which Various Decelerators Reach Mach
Targets ($M_t$) as Functions of Initial Mach Numbers
and Percent Initial $C_D A$ in VM7 Atmosphere

SECTION V - DECELERATOR SIZE

DETERMINATION

To determine the approximate sizes of expandable
decelerators needed for specified target Mach
numbers and altitudes, Goodyear Aerospace used
graphic analysis of point-mass trajectory compu-
tations.

Figure 51 schematically illustrates six steps in-
volved in this procedure. Figures 51 through 66
are curves used to determine decelerator sizes
shown in Section VI.
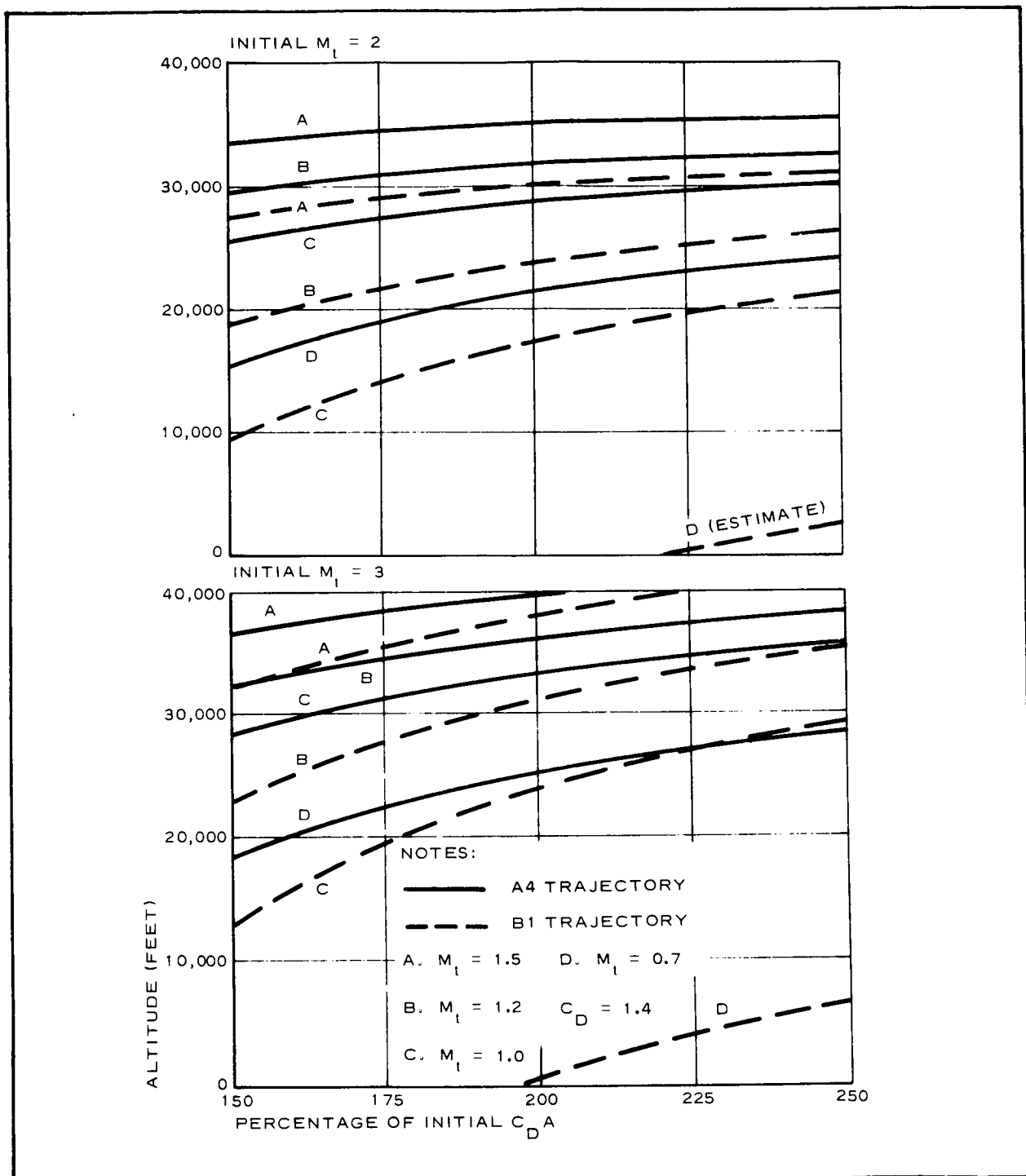
Figure 53 - Altitudes at which B1  and A4  Trajectory Decelerators
Reach Mach Targets (M_t) (VM7 Atmosphere)

NOTES:

——— A4 TRAJECTORIES

— — — B1 TRAJECTORIES

A. $M_t = 1.5$

B. $M_t = 1.2$

C. $M_t = 1.0$

D. $M_t = 0.7$

INITIAL $M_t = 5$        $C_D = 1.4$
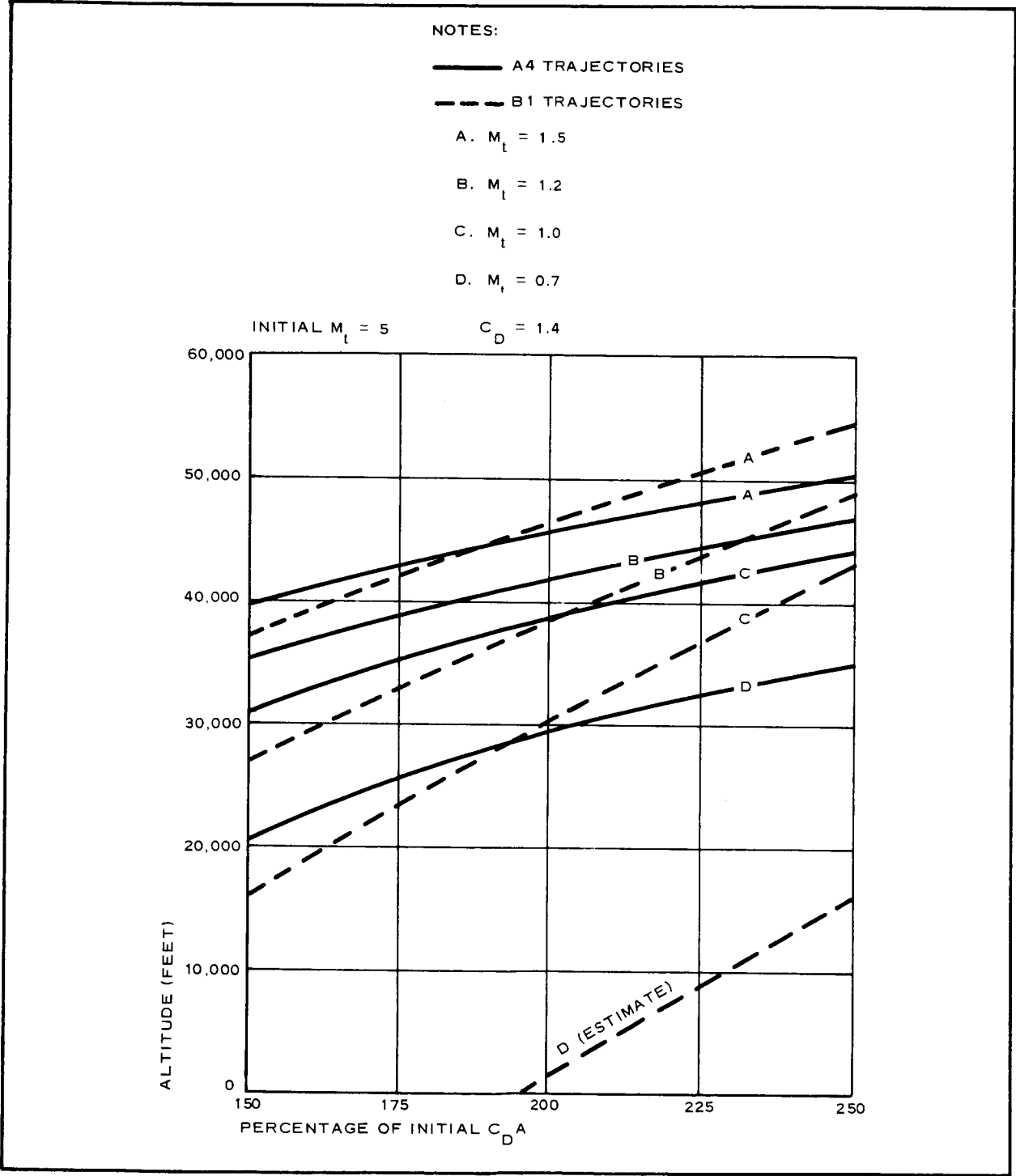
Figure 54 - Altitudes at which B1 and A4 Trajectory Decelerators
Reach Mach Targets ($M_t$) in VM7 Atmosphere
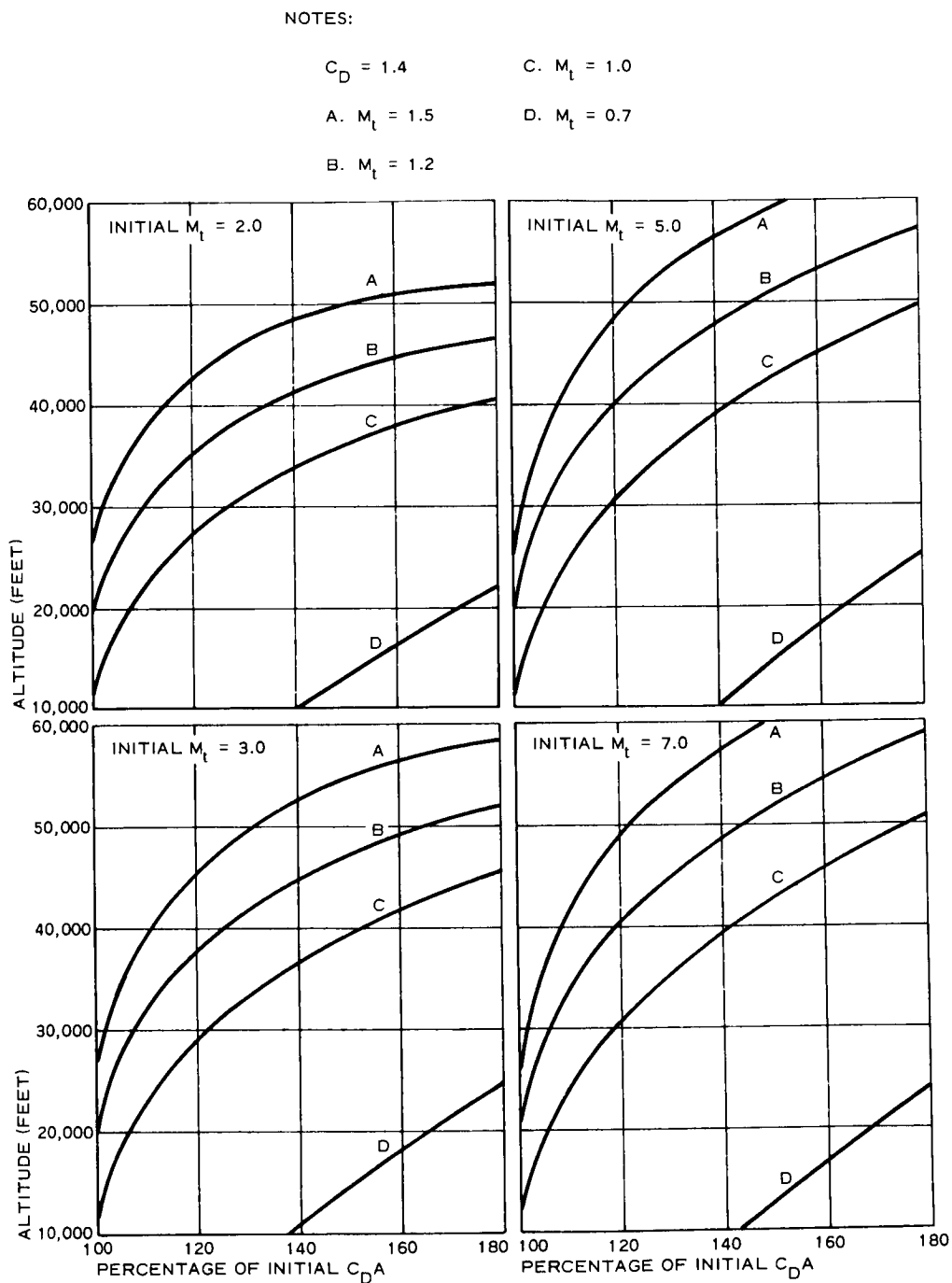
Figure 55 - Altitudes at which Trajectory 22 Decelerators
Reach Mach Targets in VM7 Atmosphere
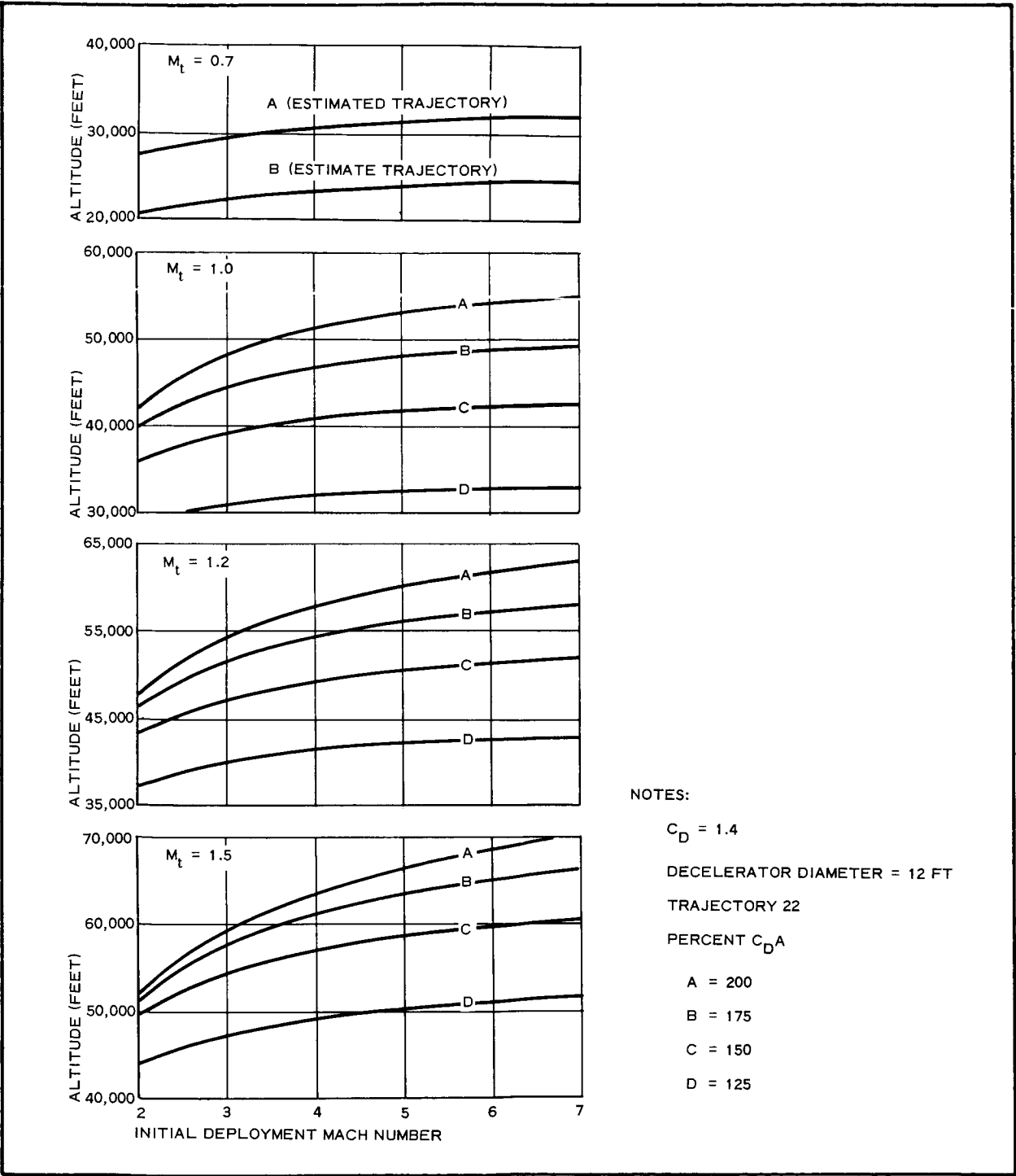
Figure 56 - Altitudes at which Trajectory 22 Decelerators Reach Mach
Targets as Functions of Initial Mach Number and Percent
Initial $C_DA$ in VM7 Atmosphere
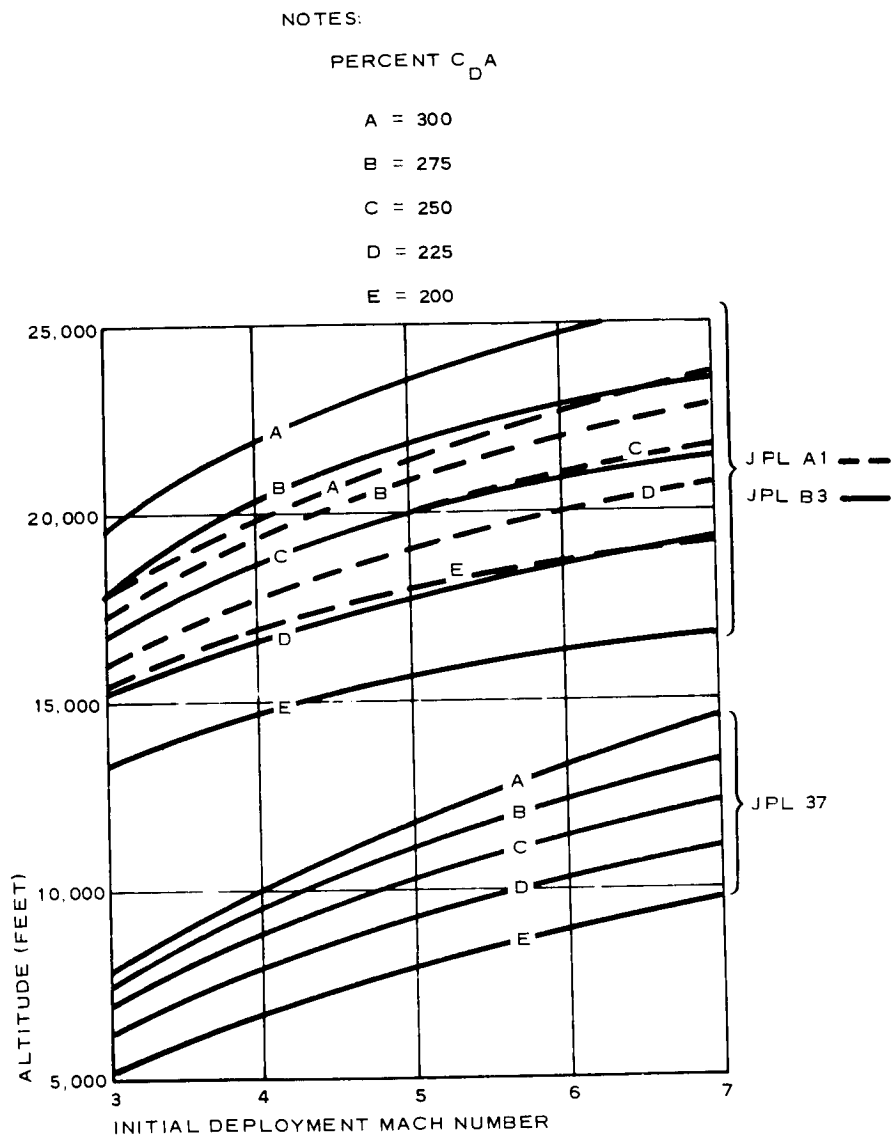
Figure 57 - Altitudes at which Various Decelerators Reach $M_t = 1.0$ as Functions of Original Mach Numbers and Percent of Initial $C_DA$ in VM8 Atmosphere
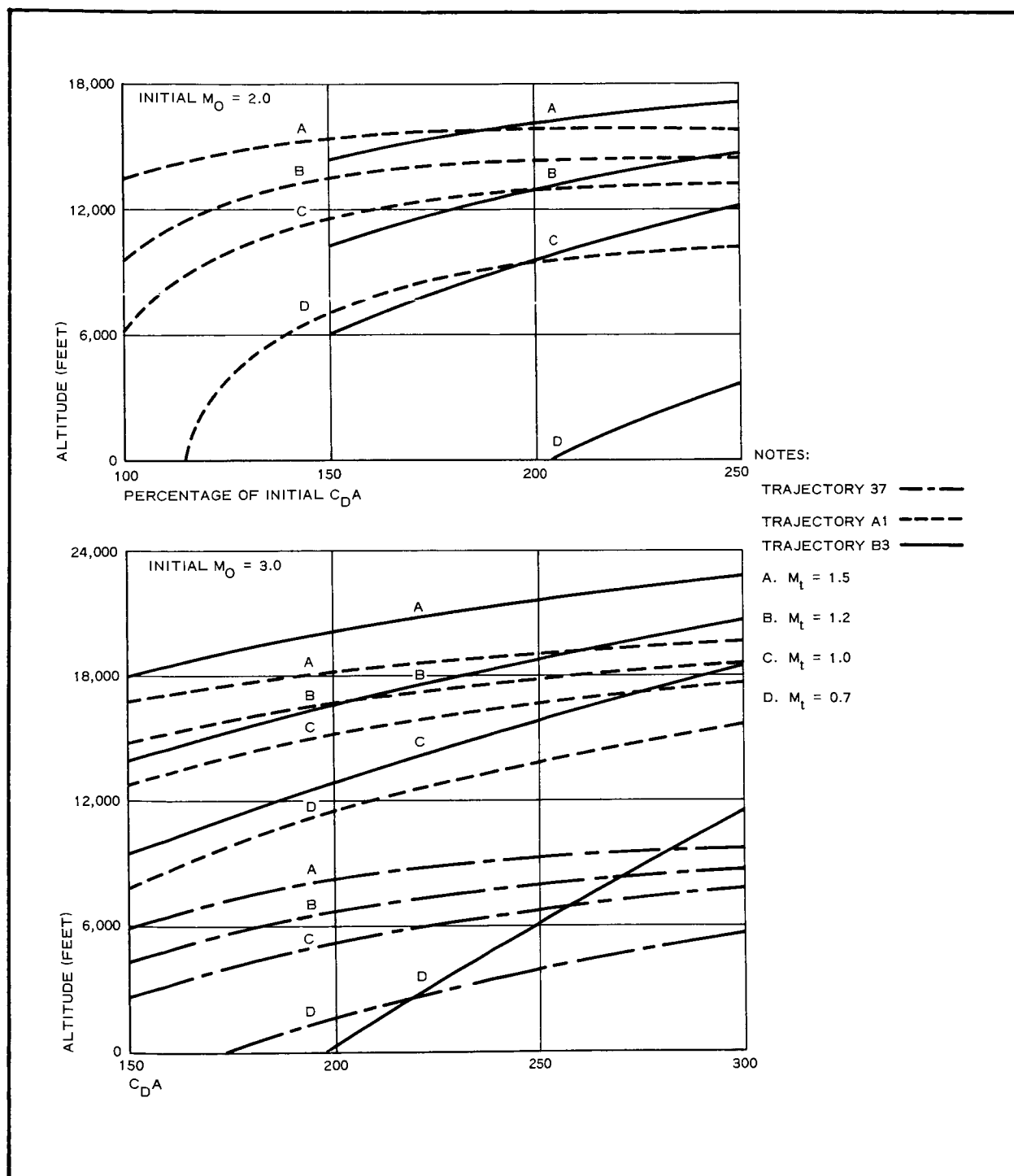
Figure 58 - Altitudes at which Decelerators Reach Mach Targets ($M_t$) in VM8 Atmosphere in A1, B3, and 37 Trajectories
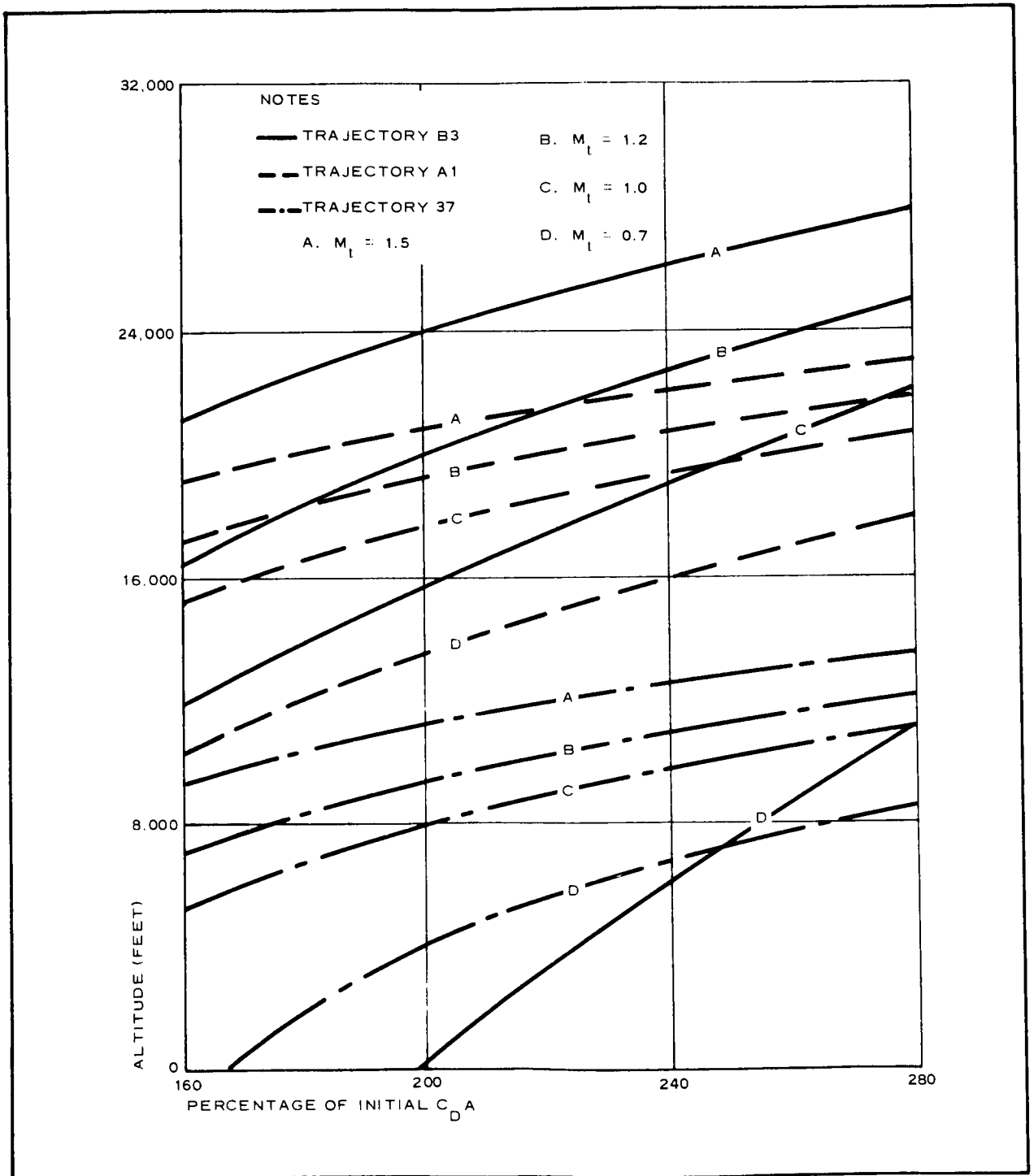
Figure 59 - Altitudes at which Decelerators Reach Mach Targets ($M_t$) in VM8 Atmosphere at Trajectories A1, B3, and 37 with Initial Mach 5.0
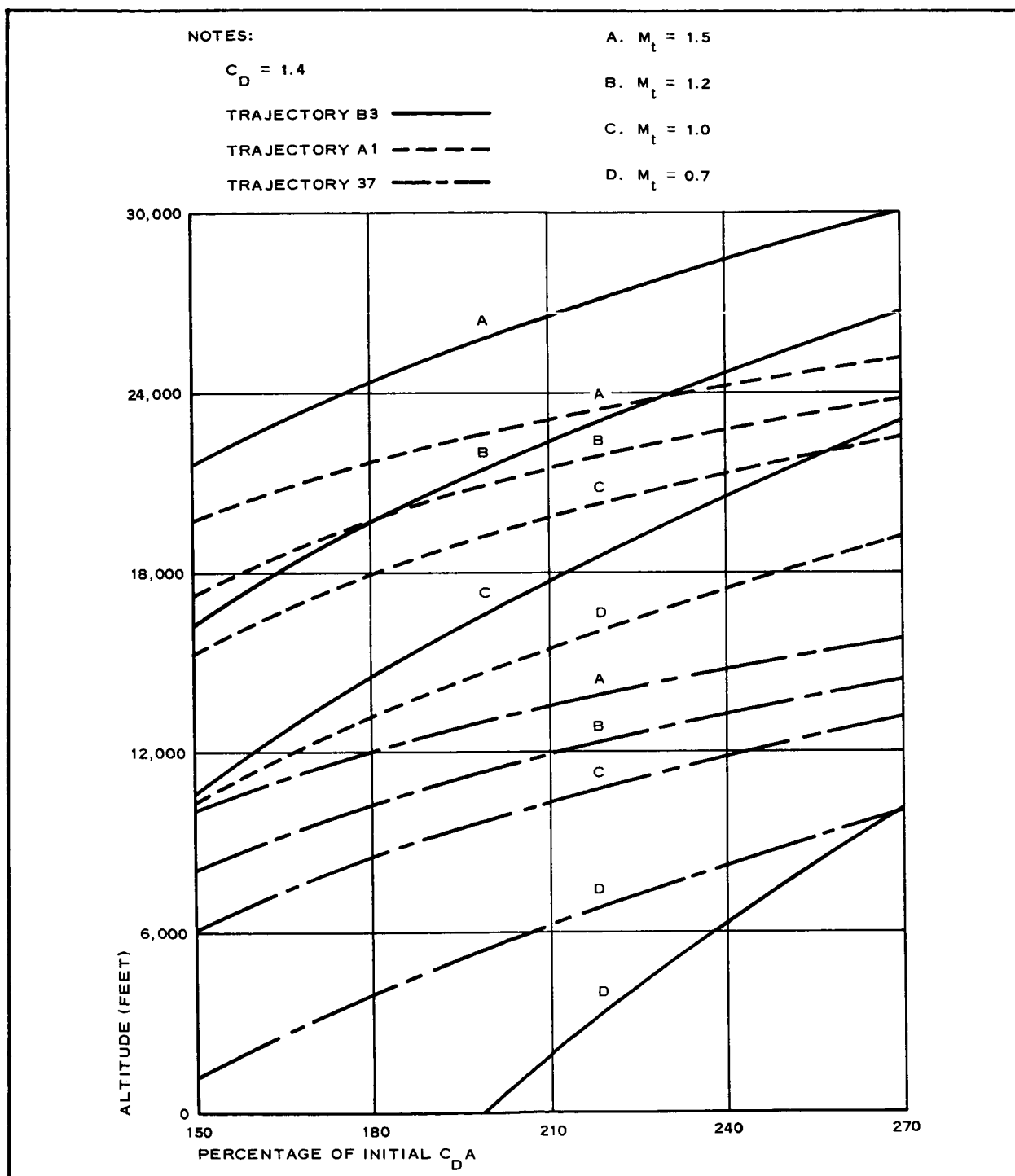
Figure 60 - Altitudes at which Decelerators Reach Mach Targets ($M_t$) in VM8 Atmosphere at Trajectories A1, B3, and 37 with Initial Mach 7.0
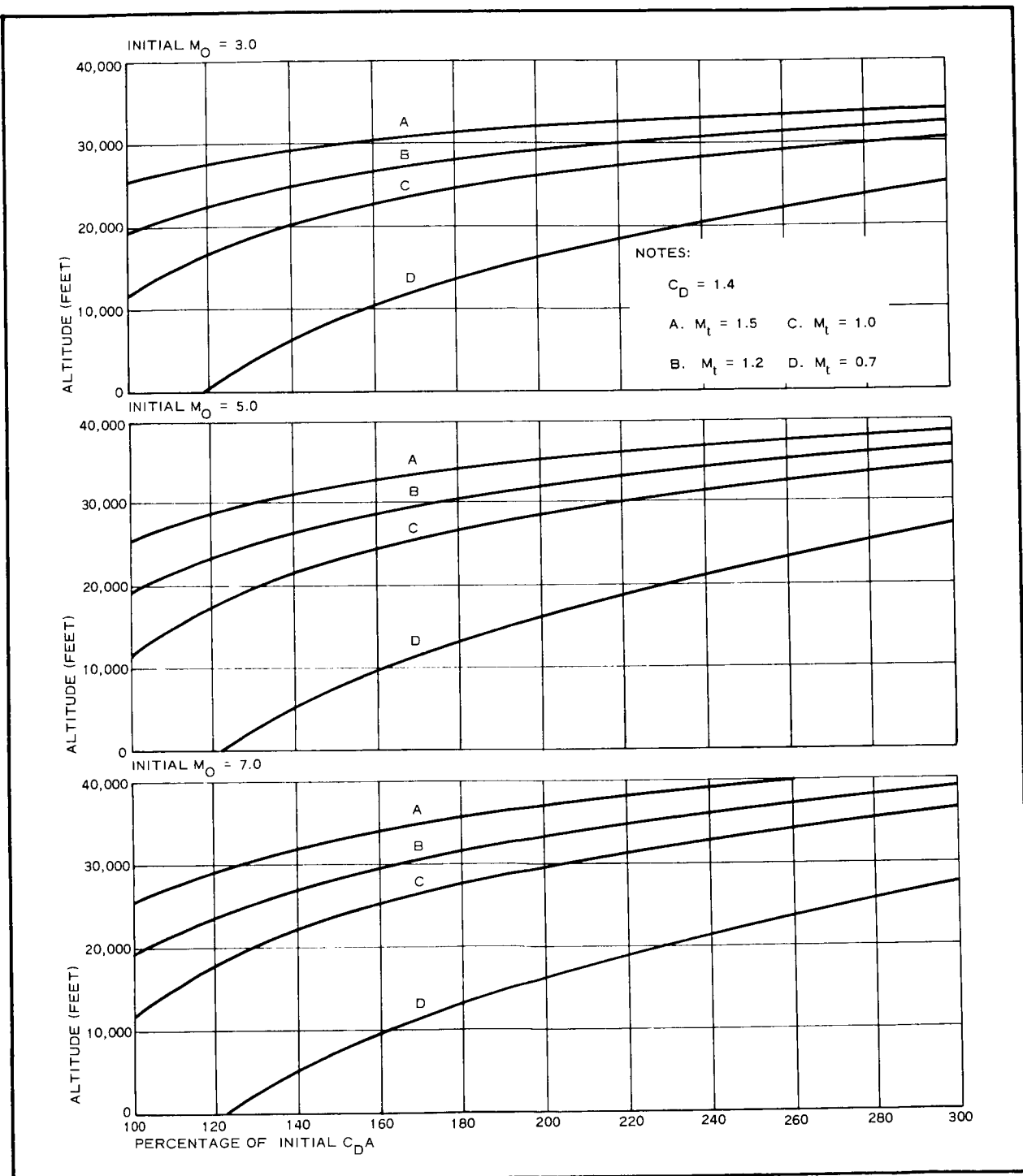
Figure 61 - Altitudes at which Decelerators Reach Mach Targets ($M_t$) in VM8 Atmosphere at Trajectory 19
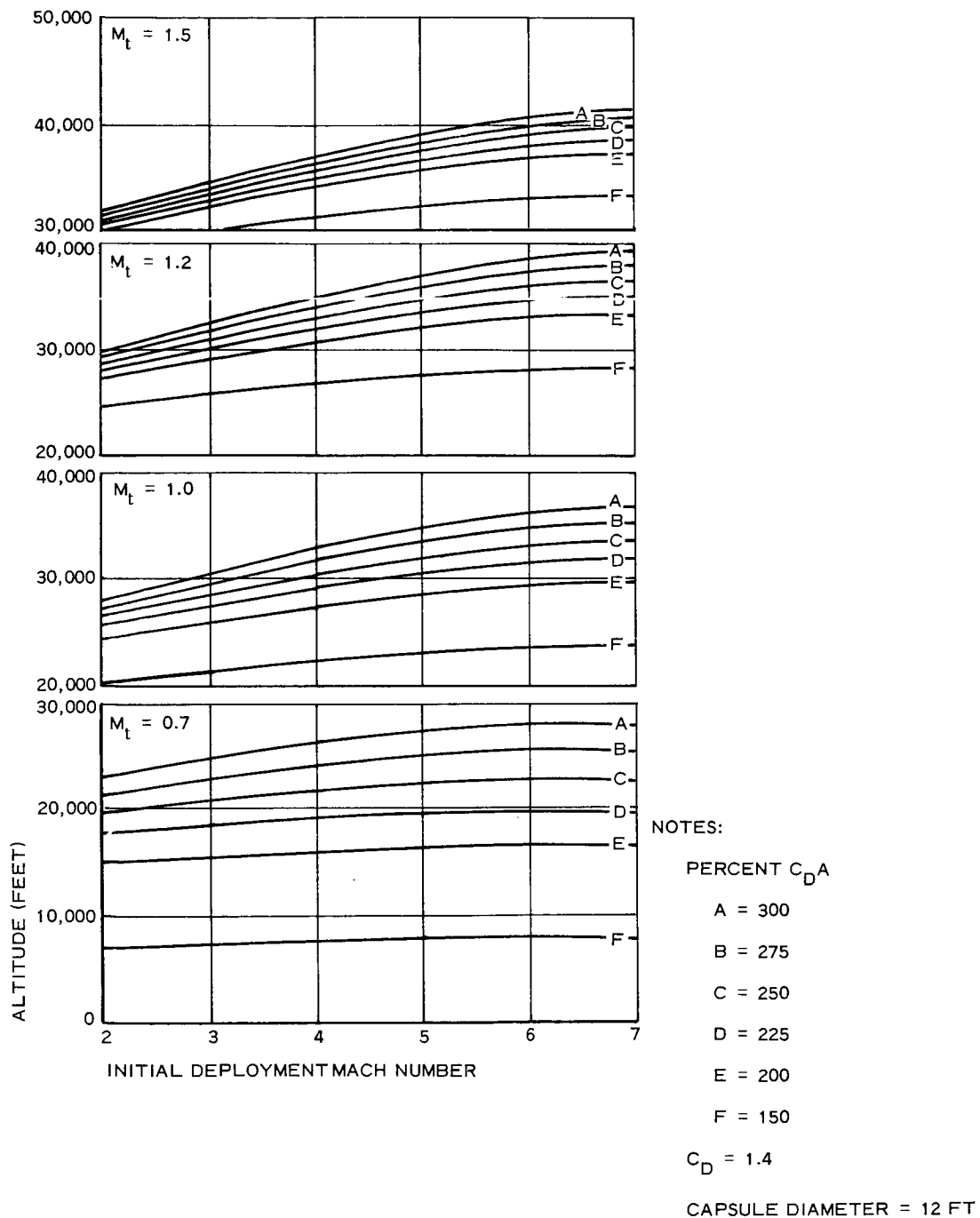
Figure 62 - Altitudes at which Decelerators Reach Mach Targets as Functions of Initial Mach Numbers and Percent of Initial $C_D A$ in VM8 Atmosphere with JPL-19 Trajectory
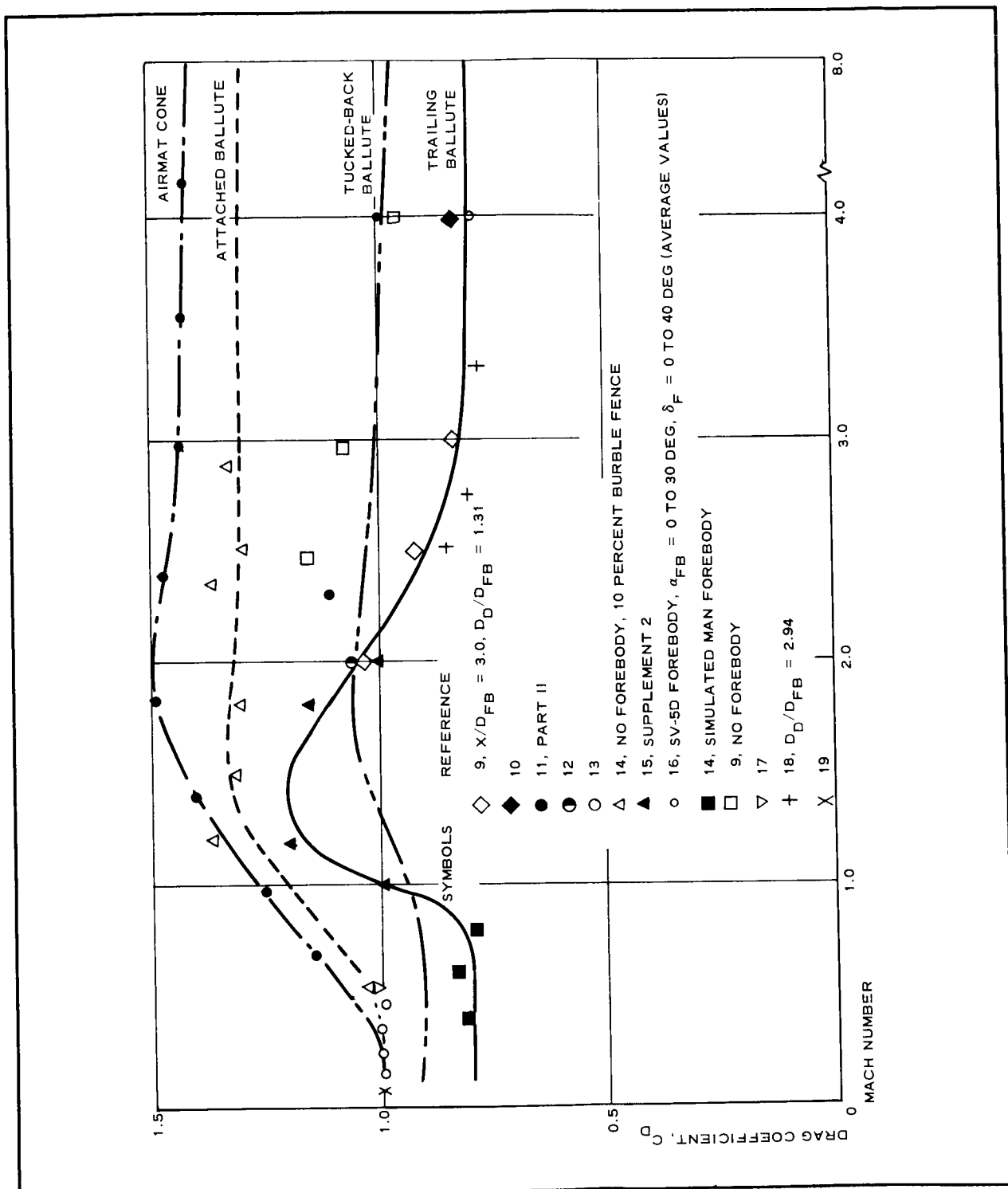
Figure 63 - Drag Coefficient Variation with Mach Number

| CONFIGURATION NUMBER | DIAMETER (FEET) | MASS (SLUGS) | $(C_D A)$ BASIC (SQUARE FEET) |
|---|---|---|---|
| A | 18.5 | 94.0 | 376.0 |
| B | 12.0 | 79.0 | 158.0 |
| C | 16.0 | 84.5 | 281.5 |
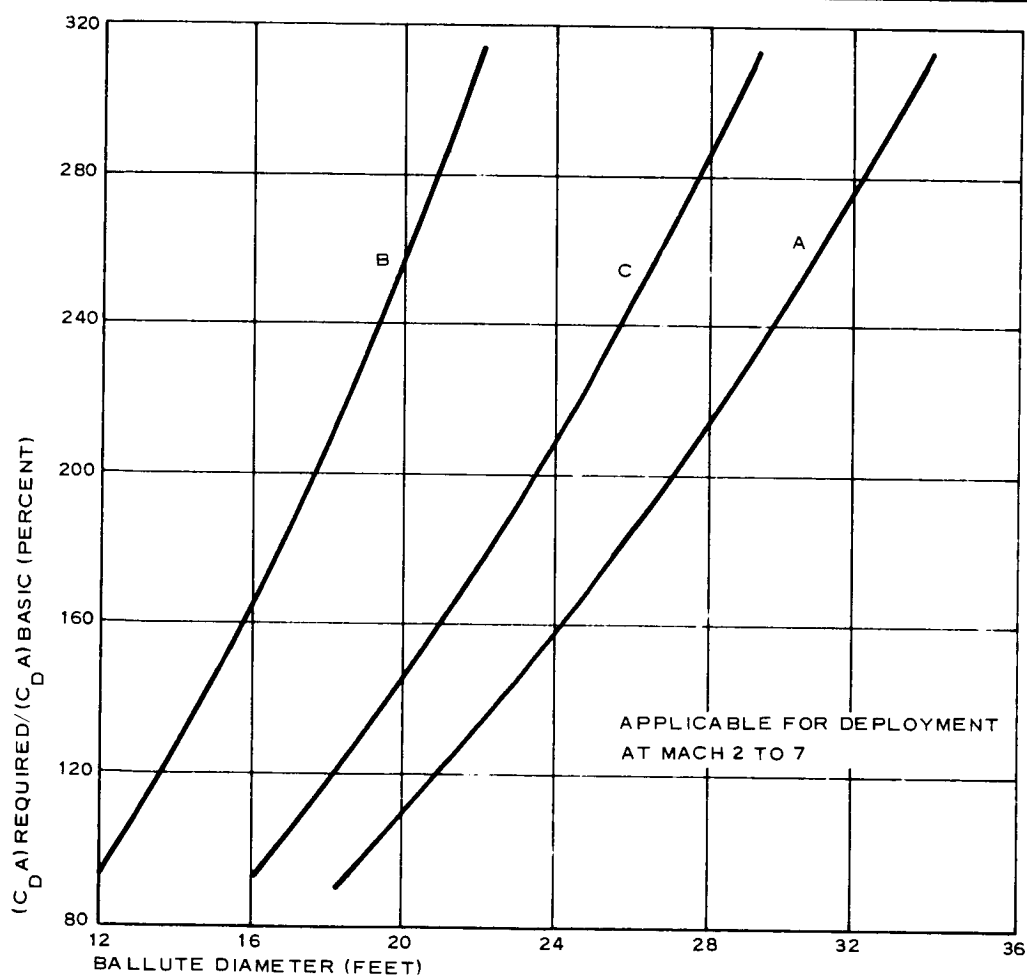


Figure 64 - Percent Drag Area versus Diameter for BALLUTE
Decelerator Applied to Three Entry Configuration

| CONFIGURATION NUMBER | DIAMETER (FEET) | MASS (SLUGS) | $(C_DA)$ BASIC (SQUARE FEET) |
|---|---|---|---|
| A | 18.5 | 94.0 | 376.0 |
| B | 12.0 | 79.0 | 158.0 |
| C | 16.0 | 84.5 | 281.5 |
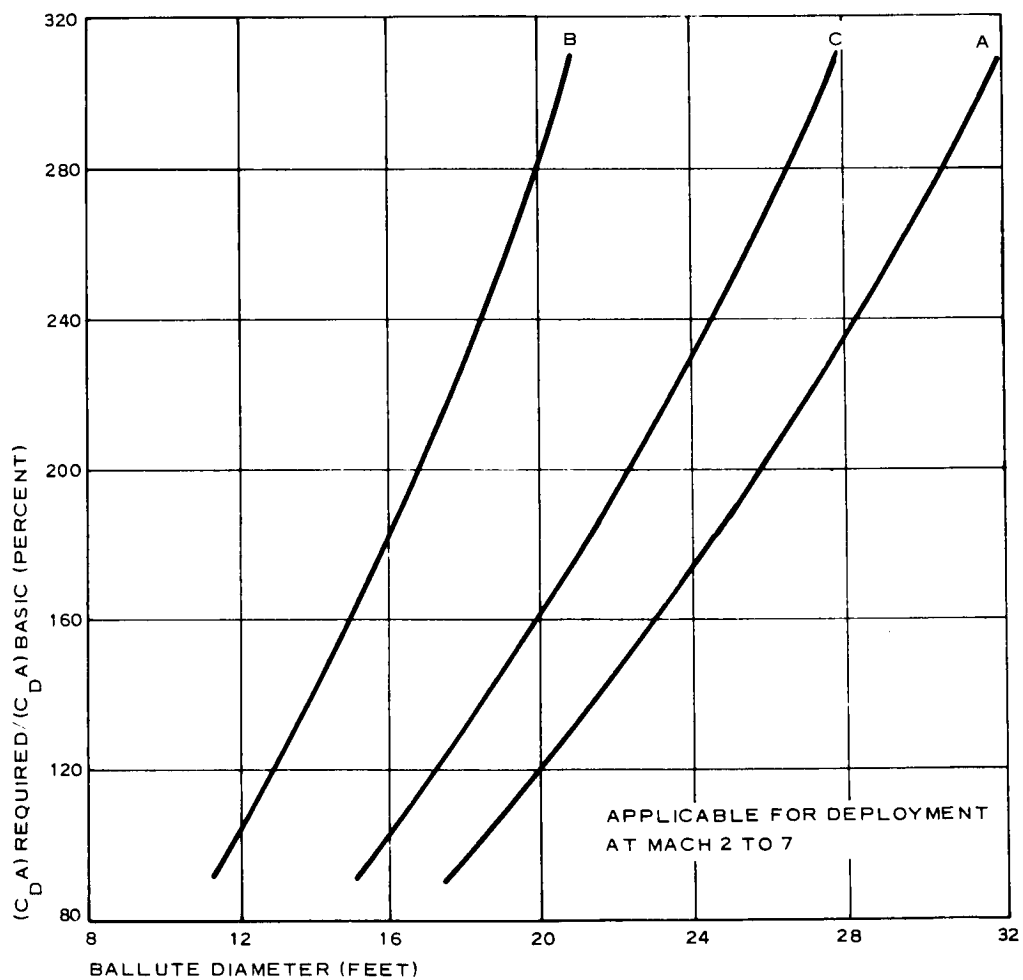


Figure 65 - Percent Drag Area versus Diameter for Inflatable Skirt
Decelerator Applied to Three Entry Configuration

| CONFIGURATION NUMBER | DIAMETER (FEET) | MASS (SLUGS) | $(C_DA)$ BASIC (SQUARE FEET) |
|---|---|---|---|
| A | 18.5 | 94.0 | 376.0 |
| B | 12.0 | 79.0 | 158.0 |
| C | 16.0 | 84.5 | 281.5 |
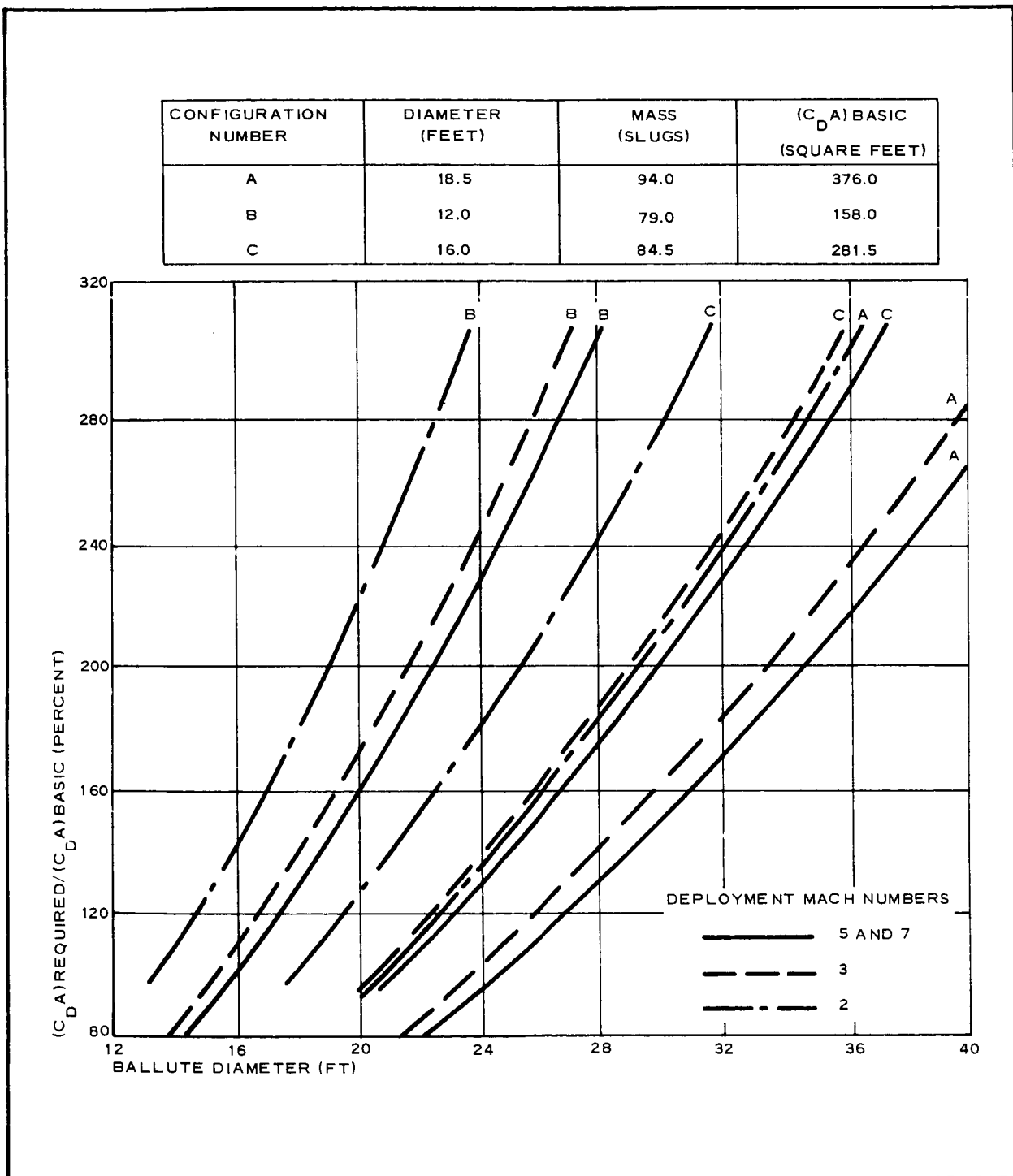


Figure 66 - Drag Area versus Diameter of Trailing BALLUTE for
Three Entry Capsule Configuration

# SECTION VI - ANALYSIS RESULTS

This section presents graphically the first approximations of decelerator size requirements determined by analyses described in Section V.

Percentage weight fractions are included for these decelerator sizes. These fractions were developed by analyses described in Appendix A of this volume as a function of the initial operating Mach number.

Figures 67 through 99 present curves that establish the minimum required decelerator envelope unit weight compatible simultaneously with aerodynamic heating and loading environments for several entry cases.

Each curve is associated with a decelerator configuration. These configurations decelerate the entry capsule to Mach 1 near the target altitude specified in the illustration. Indicated on the illustrations are optimum initial operating Mach numbers, which are associated with minimum percentage of accelerator to total entry vehicle weight.

Minimum practical envelope unit weight also is recorded on the curve illustrations for each decelerator type.

Thermal requirements are limited to a maximum temperature of 450 F for dacron and 700 F for Nomex. The minimum practical unit weight includes an 0.01-psf Neoprene or Vitron coating to provide the envelope with low porosity (approximately 0.02 cu ft/ft$^2$/sec). For this study, the coating was assumed to have the same specific heat as the envelope material.

NOTES:

    TRAJECTORY:  A1

    ATMOSPHERIC PROFILE:  VM8

    TARGET MACH NUMBER:  1.0

    TARGET ALTITUDE:  20,000 FT

    $\phi$ = OPTIMUM MACH NUMBER

ENTRY CAPSULE DATA

    DIAMETER:  18.5 FT

    MASS (SLUGS):  94

    $(C_D A)_V$ :  376 SQ FT

    $W_{EARTH}$:  3025 LB

TRAILING BALLUTE

FLARED SKIRT

ATTACHED BALLUTE

TUCKED-BACK BALLUTE

DECELERATOR DIAMETER (FEET)

INITIAL MACH NUMBER

Figure 67 - Decelerator Diameter versus Deployment Mach Number for 18.5-Ft-
Diameter Entry Capsule, Trajectory A1

NOTES.

TRAJECTORY DESIGNATION: A1          MASS (SLUGS): 94

ATMOSPHERIC PROFILE: VM8          $(C_D A)_V$     376 SQ FT

TARGET MACH NUMBER: 1.0

TARGET ALTITUDE: 20,000 FT          $W_{EARTH}$: 3025 LB

$\phi$ = OPTIMUM MACH NUMBER

A = NOMEX AT 600 F

B = DACRON AT 350 F

ENTRY CAPSULE DATA

DIAMETER. 18.5 FT

Figure 68 - Percent Decelerator to Total System Weight versus Deployment Mach Number for 18.5-Ft-Diameter Entry Capsule, Trajectory Al

NOTES:   TRAJECTORY DESIGNATION: B3      ENTRY CAPSULE DATA:

ATMOSPHERIC PROFILE: VM8          DIAMETER: 12 FT

TARGET MACH NUMBER: 1.0           MASS (SLUGS): 79

TARGET ALTITUDE: 20,000 FT        $(C_D A)_V$ : 158 SQ FT

$\phi$ = OPTIMUM MACH NUMBER       $W_{EARTH}$: 2540 LB

Figure 69 - Decelerator Diameter versus Deployment Mach Number for 12-Ft-Diameter Entry Capsule, Trajectory B3

Figure 70 - Percent Decelerator to Total System Weight versus Deployment
Mach Number for 12-Ft-Diameter Entry Capsule, Trajectory B3

NOTES:

TRAJECTORY DESIGNATION: 37          ENTRY CAPSULE DATA:

ATMOSPHERIC PROFILE: VM8               DIAMETER: 16 FT

TARGET MACH NUMBER: 1.0               MASS (SLUGS): 84.5

TARGET ALTITUDE: 10,000 FT            $(C_D A)_V$ : 281.5 SQ FT

$\phi$ = OPTIMUM MACH NUMBER            $W_{EARTH}$: 2720 LB



Figure 71 - Decelerator Diameter versus Deployment Mach Number for 16-Ft Entry Capsule Diameter, Trajectory 37

NOTES:

TRAJECTORY DESIGNATION: 37          ENTRY CAPSULE DATA

ATMOSPHERIC PROFILE: VM8               DIAMETER: 16 FT

TARGET MACH NUMBER: 1.0                MASS (SLUGS): 84.5

TARGET ALTITUDE: 10,000 FT             $(C_D A)_V$ : 281.5 SQ FT

$\phi$ = OPTIMUM MACH NUMBER

A = NOMEX AT 600 F                     $W_{EARTH}$: 2720 LB

B = DACRON AT 350 F

TRAILING BALLUTE

TUCKED-BACK BALLUTE

ATTACHED BALLUTE

FLARED SKIRT

DECELERATOR-TO-TOTAL WEIGHT (PERCENT)

INITIAL MACH NUMBER
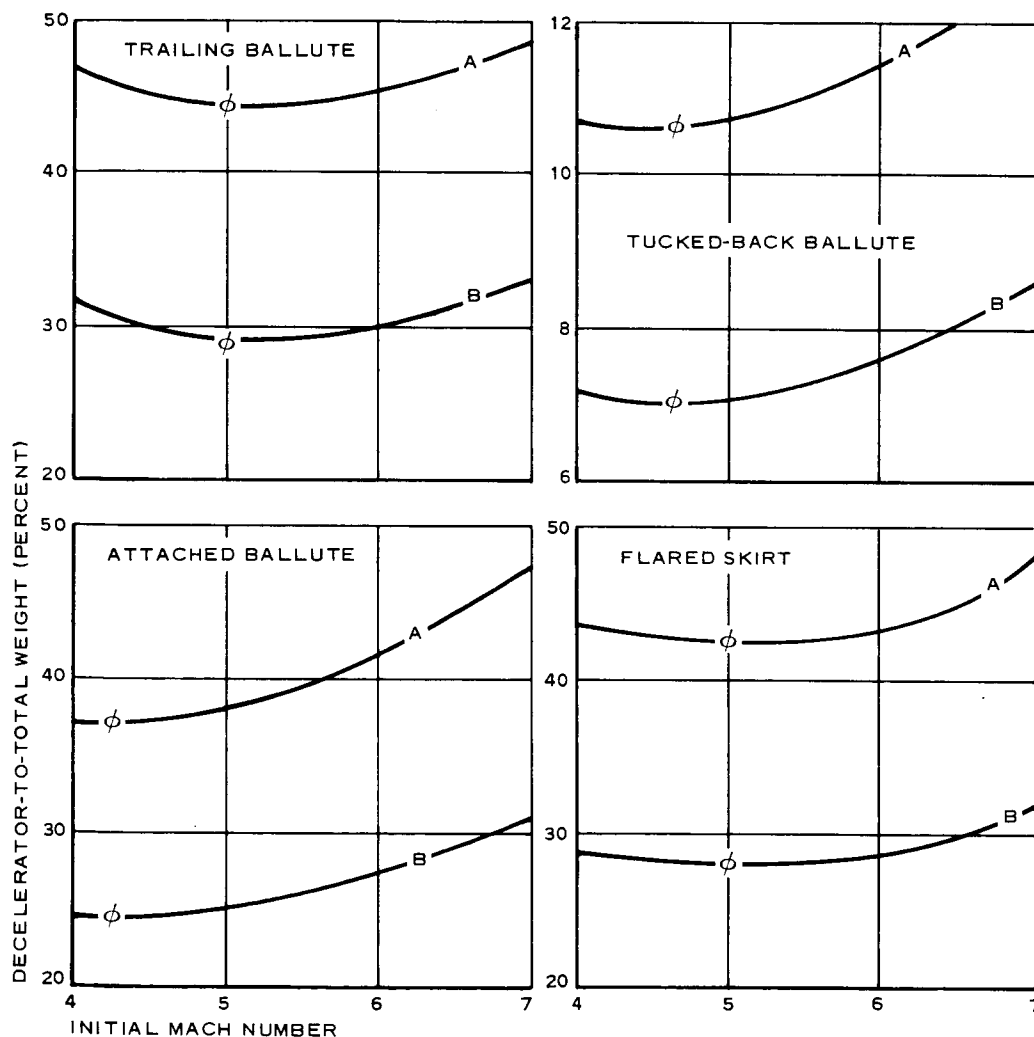
Figure 72 - Percent Decelerator to Total System Weight versus Deployment
Mach Number for 16-Ft-Diameter Entry Capsule, Trajectory 37
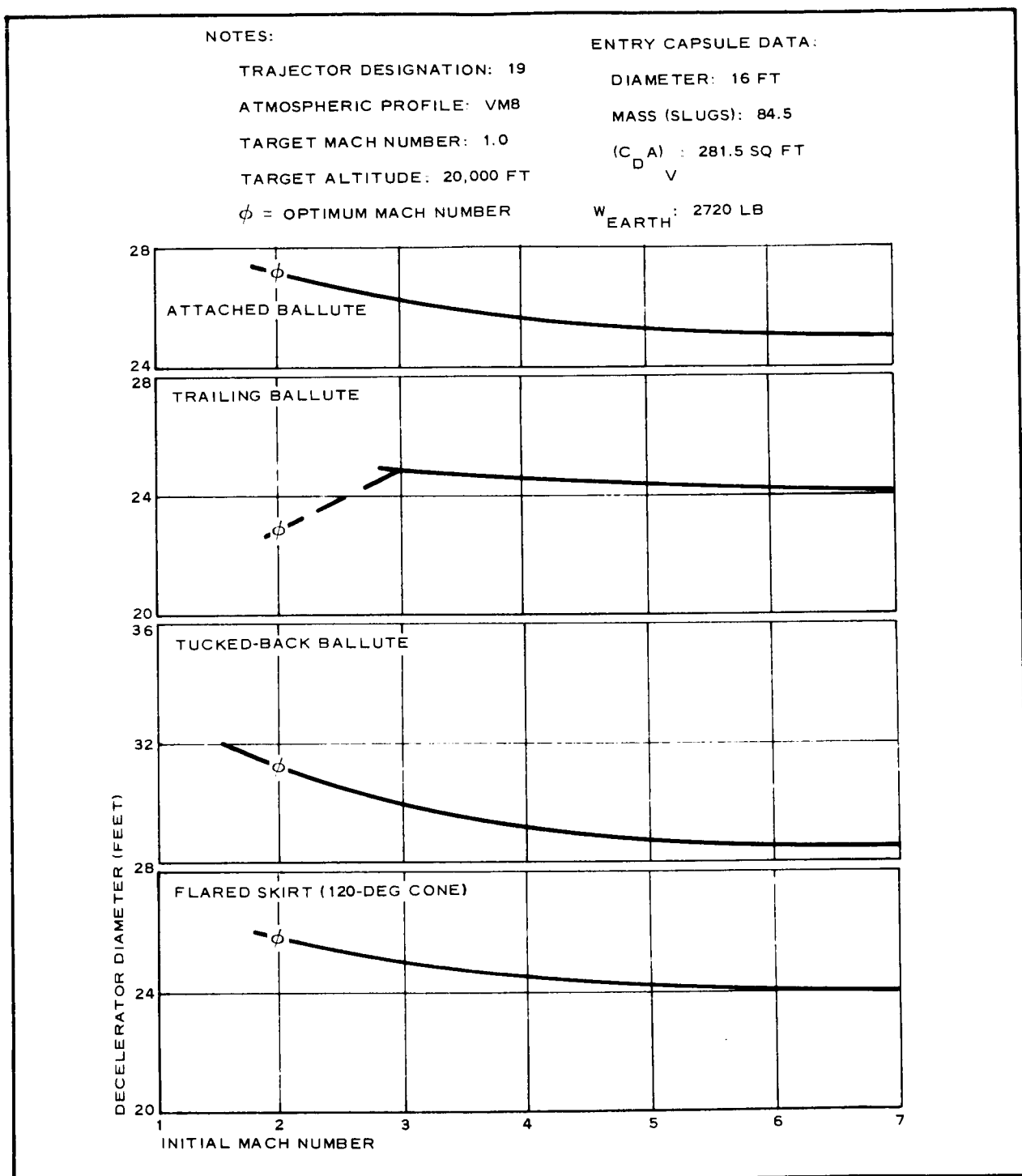
Figure 73 - Decelerator Diameter versus Deployment Mach Number for 16-Ft-
Diameter Entry Capsule, Trajectory 19 at 20,000 Ft

NOTES:

TRAJECTORY DESIGNATION: 19

ATMOSPHERIC PROFILE: VM8

TARGET MACH NUMBER: 1.0

TARGET ALTITUDE: 20,000 FT

$\phi$ = SELECTED MACH NUMBER

A = NOMEX 600 F

B = DACRON 350 F

ENTRY CAPSULE DATA:

DIAMETER: 16 FT

MASS (SLUGS): 84.5

$(C_D A)_V$ : 281.5 SQ FT

$W_{EARTH}$: 2720 LB
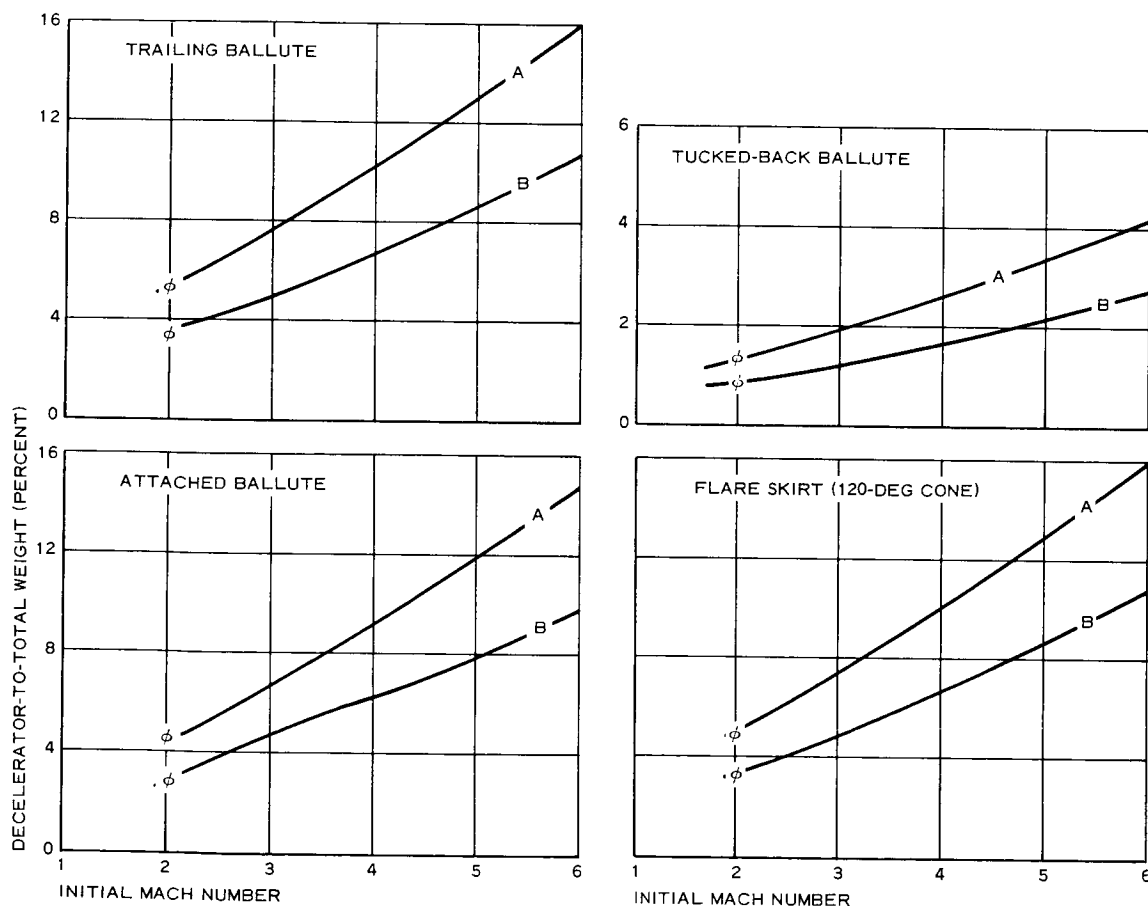


Figure 74 - Percent Decelerator to Total System Weight versus Deployment Mach Number for 16-Ft Entry Capsule, Trajectory 19 at 20,000 Ft

Figure 75 - Decelerator Diameter versus Deployment,  16-Ft Diameter
Entry Capsule,  Trajectory 19 at 30,000 Ft
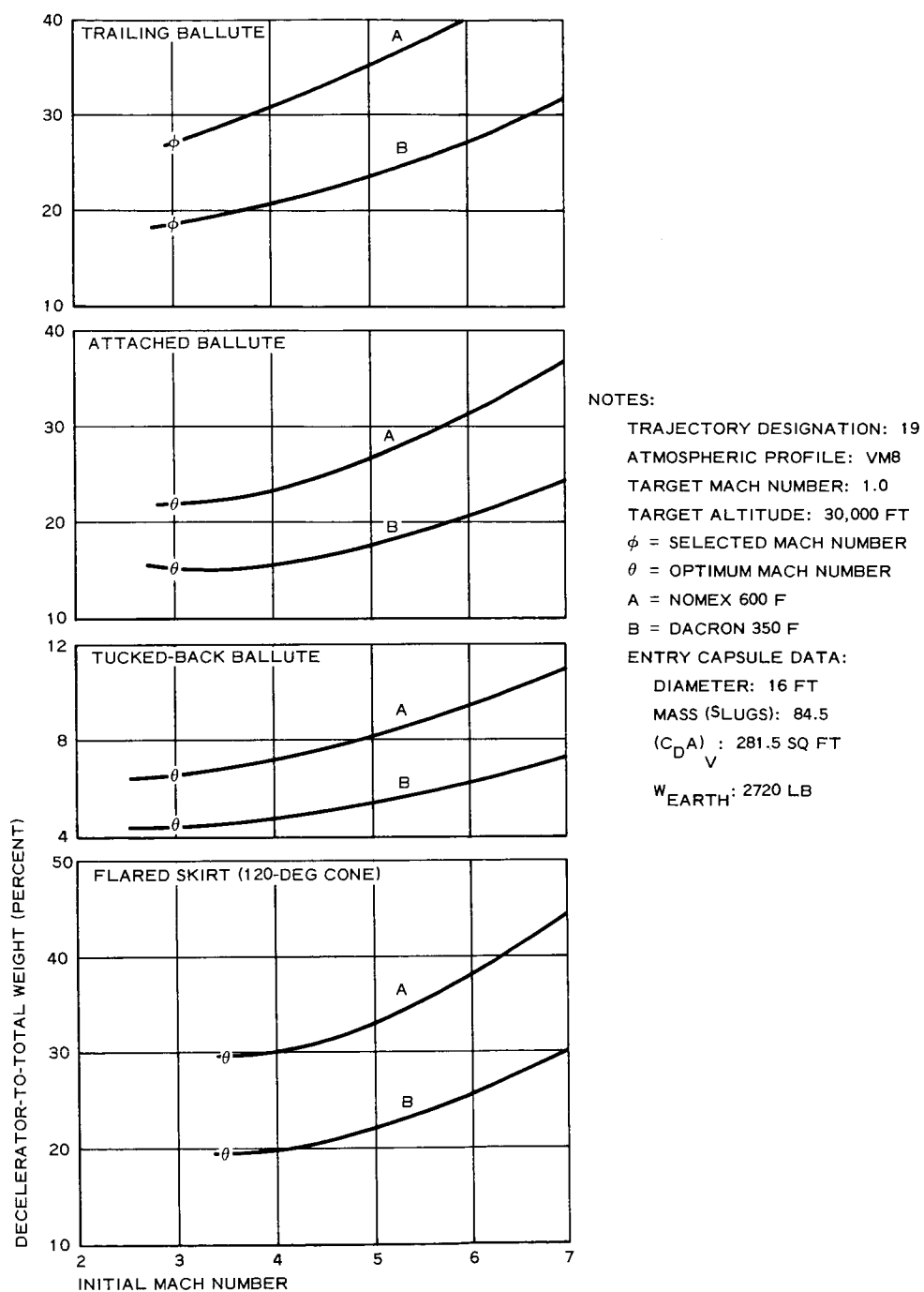
Figure 76 - Percent Decelerator to Total Weight versus Deployment Mach Number, 16-Ft-Diameter Entry Capsule Trajectory 19 at 30,000 Ft

NOTES:

TRAJECTORY DESIGNATION: A4

ATMOSPHERIC PROFILE: VM7

TARGET MACH NUMBER: 1.0

TARGET ALTITUDE: 20,000 FT

$\phi$ = SELECTED MACH NUMBER

$\theta$ = OPTIMUM MACH NUMBER

ENTRY CAPSULE DATA:

DIAMETER: 18.5 FT

MASS (SLUGS): 94

$(C_D A)_V$ : 376 SQ FT

$W_{EARTH}$: 3025 LB
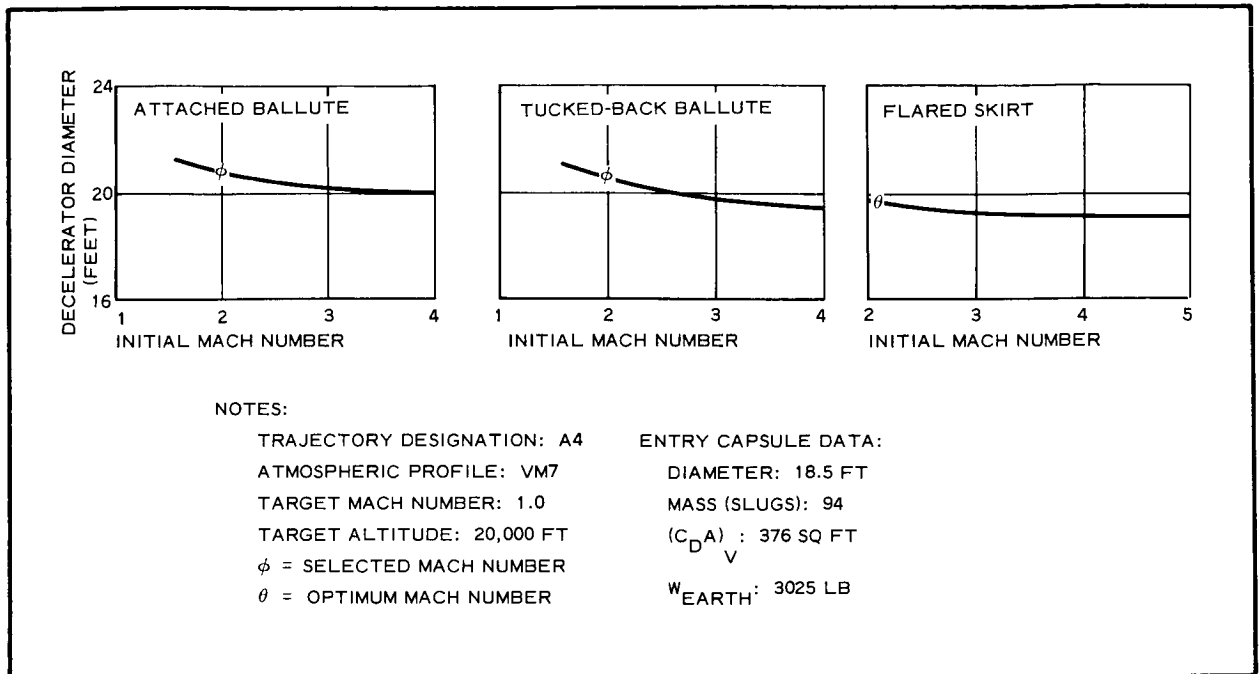
Figure 77 - Decelerator Diameter versus Deployment Mach Number for 18.5-Ft-Diameter Entry Capsule, Trajectory A4 at 20,000 Ft

NOTES:

TRAJECTORY DESIGNATION: A4

ATMOSPHERIC PROFILE: VM7

TARGET MACH NUMBER: 1.0

TARGET ALTITUDE: 20,000 FT

$\phi$ = SELECTED MACH NUMBER

$\theta$ = OPTIMUM MACH NUMBER

A = NOMEX 600 F

B = DACRON 350 F

ENTRY CAPSULE DATA:

DIAMETER: 18.5 FT

MASS (SLUGS): 94

$(C_D A)_V$ : 376 SQ FT

$W_{EARTH}$: 3025 LB
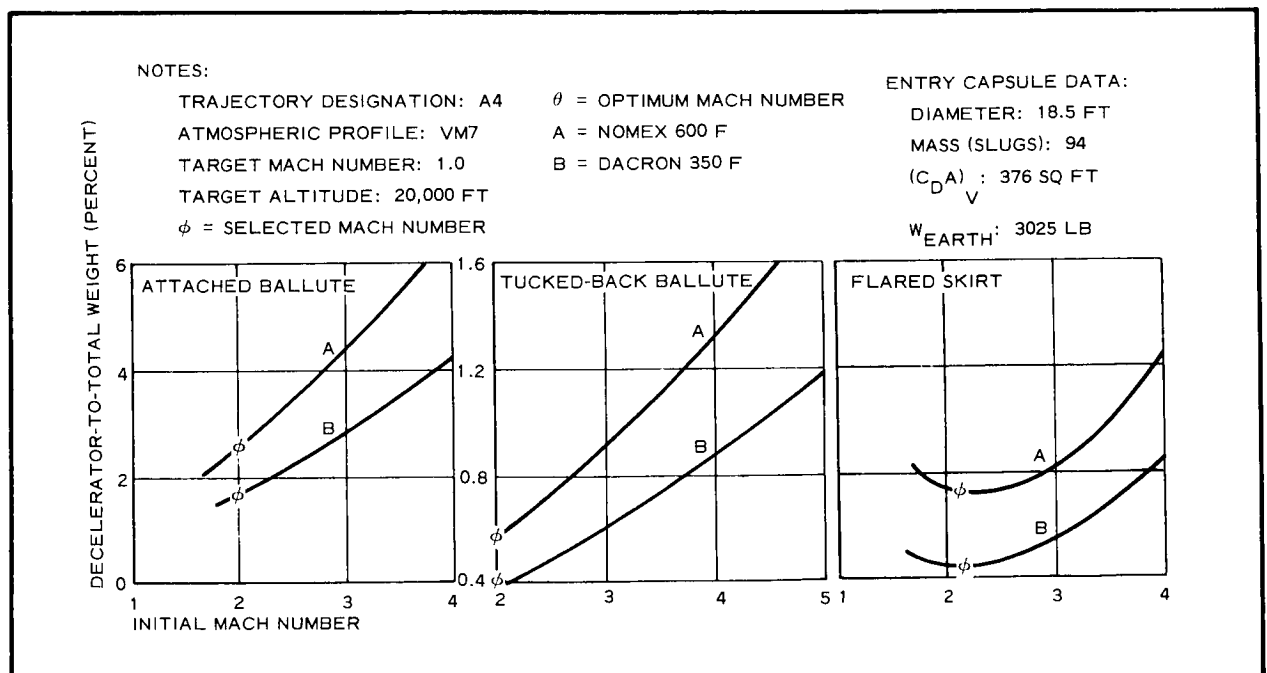
Figure 78 - Percent Decelerator to Total Weight versus Deployment Mach Number, 18.5-Ft-Diameter Entry Capsule, Trajectory A4 at 20,000 Ft
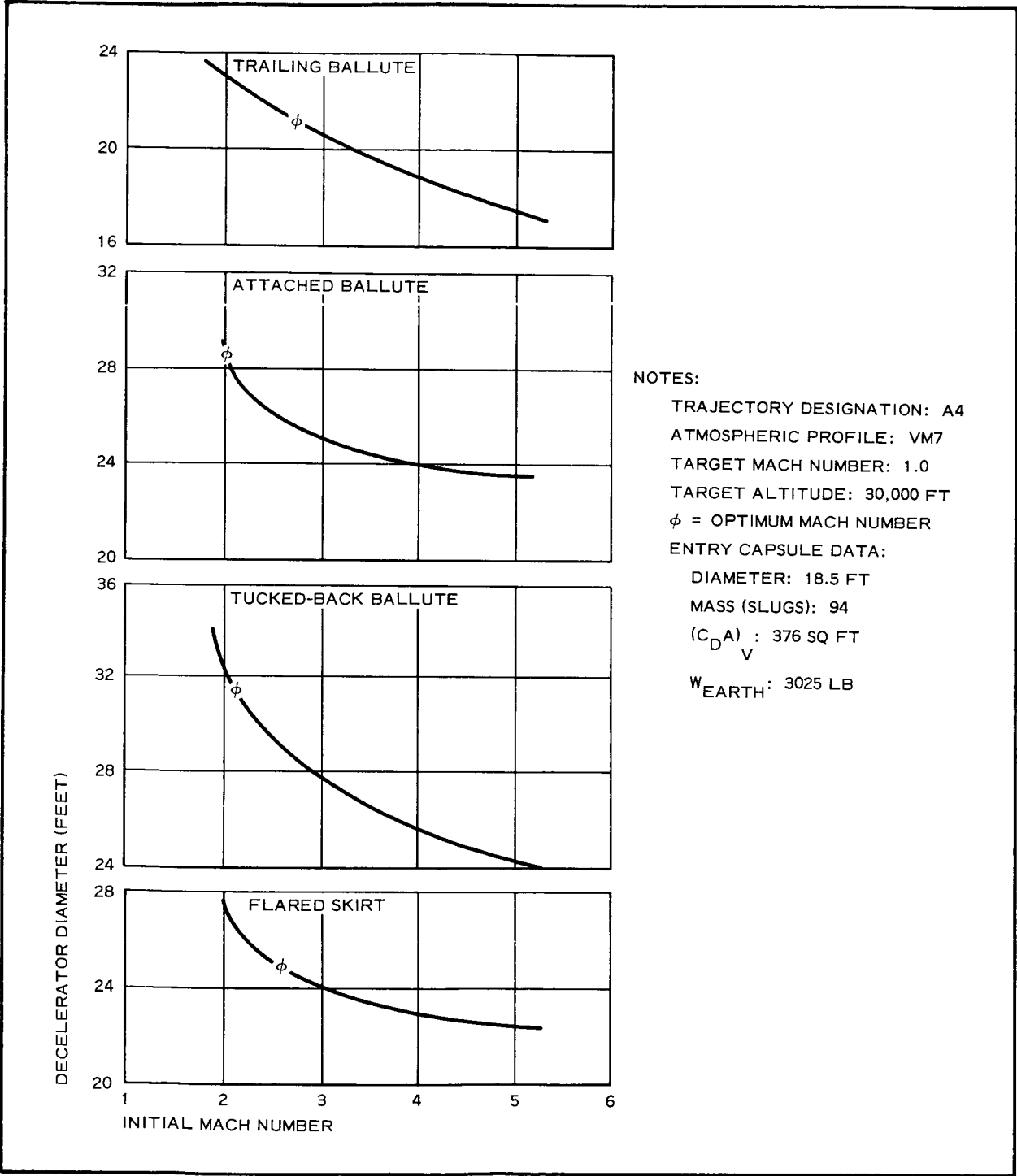
Figure 79 - Decelerator Diameter versus Deployment Mach Number,
18.5-Ft-Diameter Entry Capsule, Trajectory A4 at
30,000 Ft
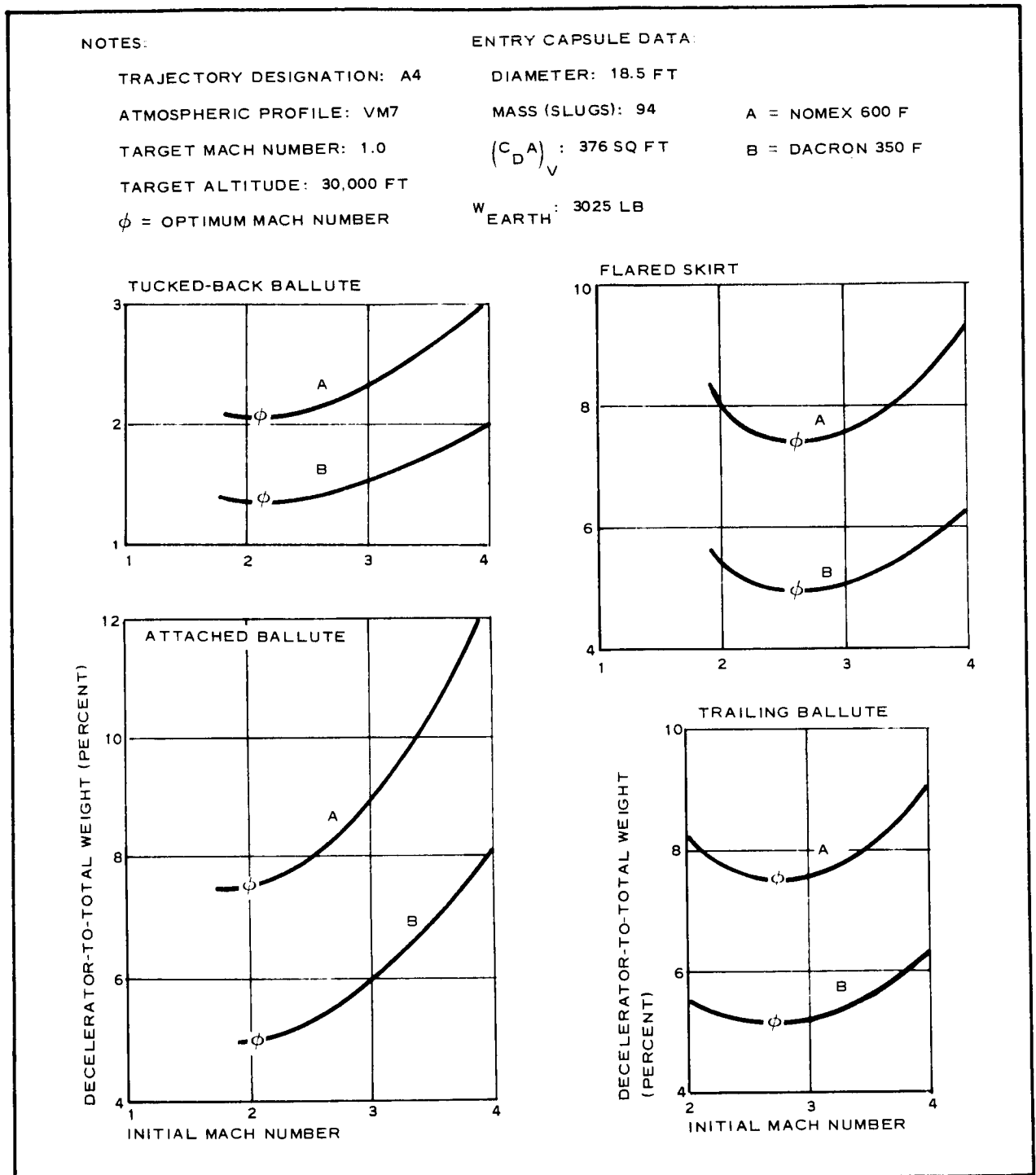
Figure 80 - Percent Decelerator to Total Weight versus Deployment Mach
Number, 18.5-Ft Entry Capsule Diameter, Trajectory A4 at
30,000 Ft

NOTES:

TRAJECTORY DESIGNATION: B1          ENTRY CAPSULE DATA:

ATMOSPHERIC PROFILE: VM7               DIAMETER: 12 FT

TARGET MACH NUMBER: 1.0                MASS (SLUGS): 79

TARGET ALTITUDE: 20,000 FT            $(C_D A)_V$ : 158 SQ FT

$\theta$ = OPTIMUM MACH NUMBER

$\phi$ = MACH NUMBER SELECTED         $W_{EARTH}$: 2540 LB

TRAILING BALLUTE

ATTACHED BALLUTE

TUCKED-BACK BALLUTE

FLARED SKIRT

DECELERATOR DIAMETER (FEET)

INITIAL MACH NUMBER

Figure 81 - Decelerator Diameter versus Deployment Mach Number, 12-Ft
Entry Capsule Diameter, Trajectory B1 at 20,000 Ft
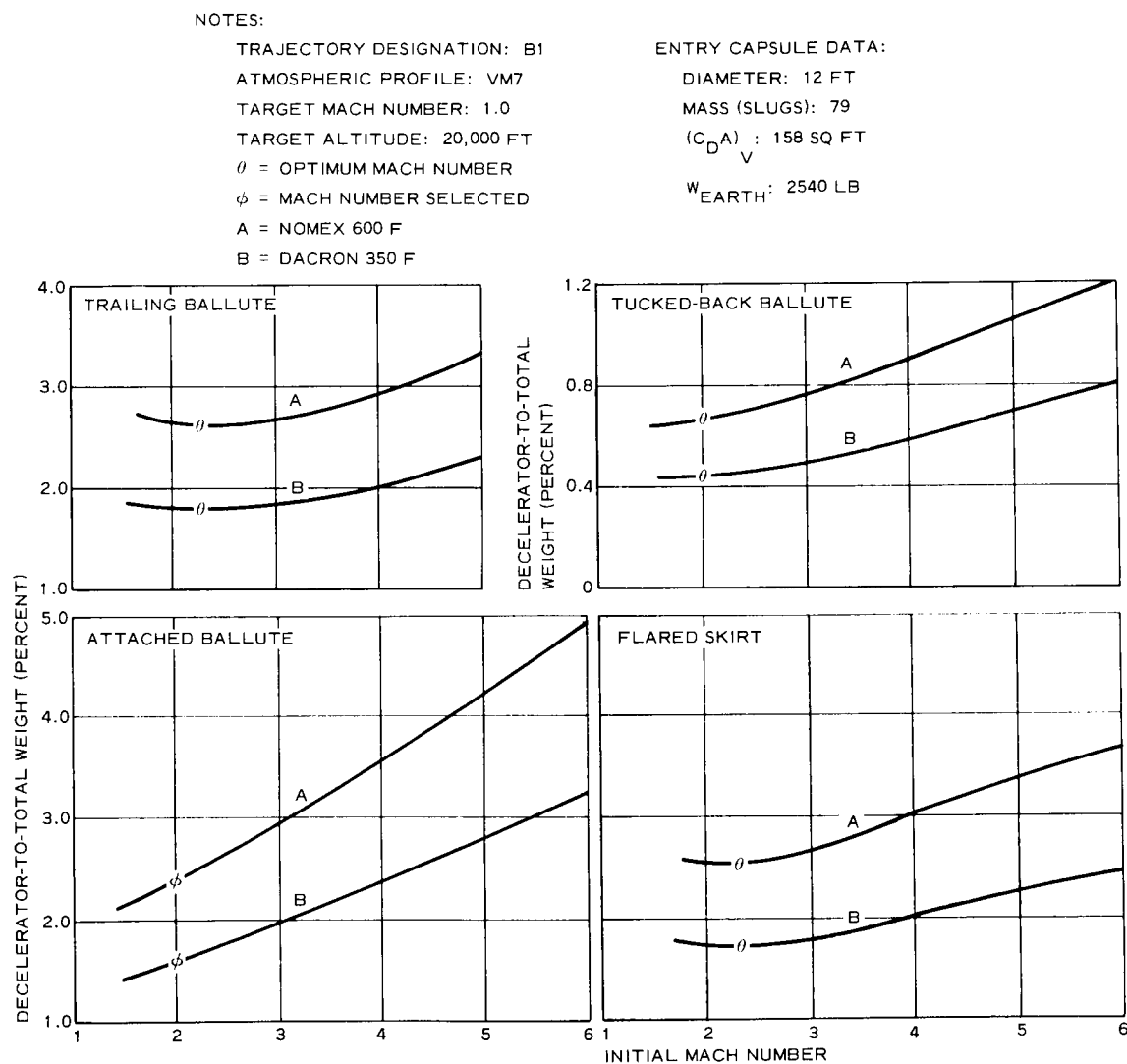
Figure 82 - Percent Decelerator to Total System Weight versus Deployment Mach Number, 12-Ft Diameter Entry Capsule, Trajectory B1 at 20,000 Ft

NOTES:

    TRAJECTORY DESIGNATION: B1

    ATMOSPHERIC PROFILE: VM7

    TARGET MACH NUMBER: 1.0

    TARGET ALTITUDE: 30,000 FT

    $\phi$ = SELECTED MACH NUMBER

ENTRY CAPSULE DATA:

    DIAMETER: 12 FT

    MASS (SLUGS): 79

    $(C_D A)_V$ : 158 SQ FT

    $W_{EARTH}$: 2540 LB
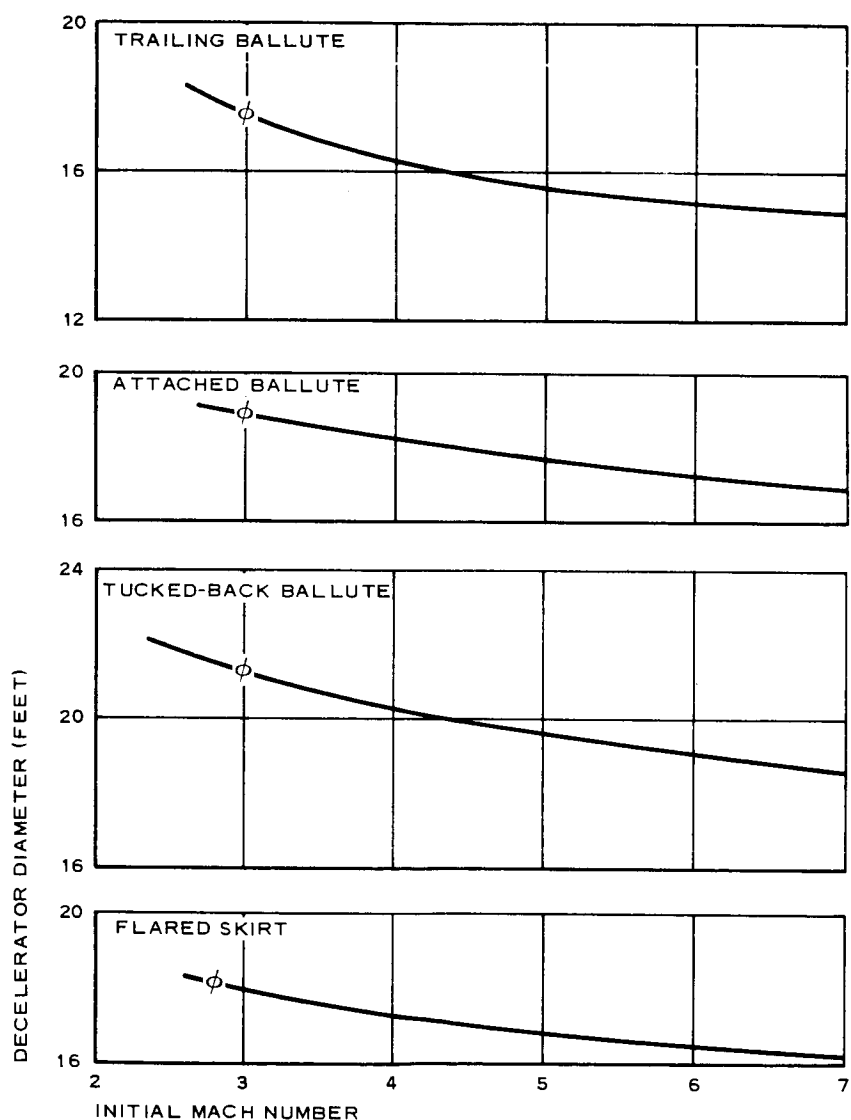
Figure 83 - Decelerator Diameter versus Deployment Mach Number for 12-Ft Diameter Entry Capsule, Trajectory B1 at 30,000 Ft
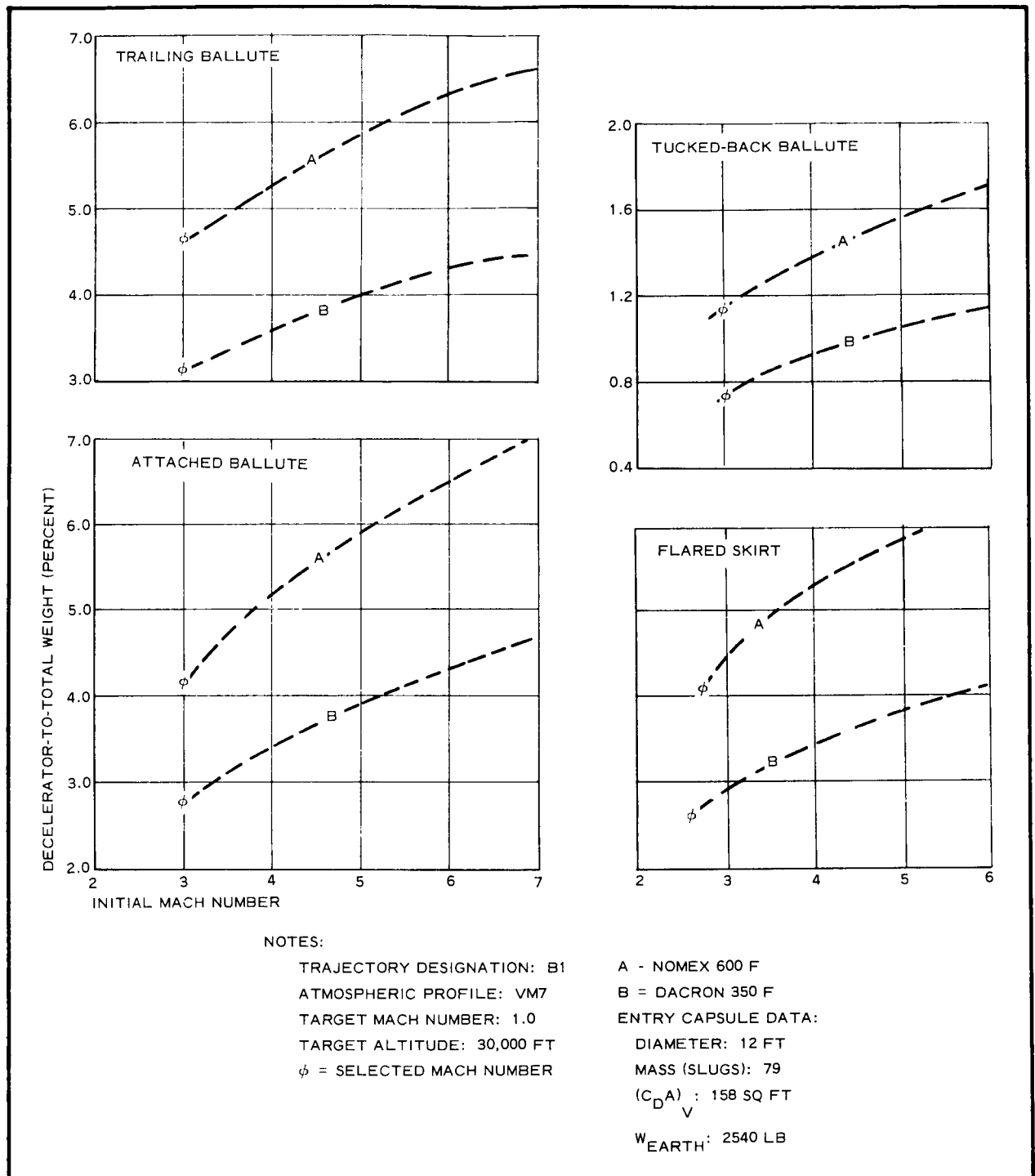
Figure 84 - Percent Decelerator to Total Weight versus Deployment Mach
Number for 12-Ft Entry Capsule, Trajectory B1 at 30,000 Ft

NOTES:

TRAJECTORY DESIGNATION: 22

ATMOSPHERIC PROFILE: VM7

TARGET MACH NUMBER: 1.0

TARGET ALTITUDE: 20,000 FT

$\phi$ = SELECTED MACH NUMBER

ENTRY CAPSULE DATA:

DIAMETER: 16 FT

MASS (SLUGS): 84.5

$\left(C_D A\right)_V$ : 281.5 SW FT

$W_{EARTH}$: 2720 LB
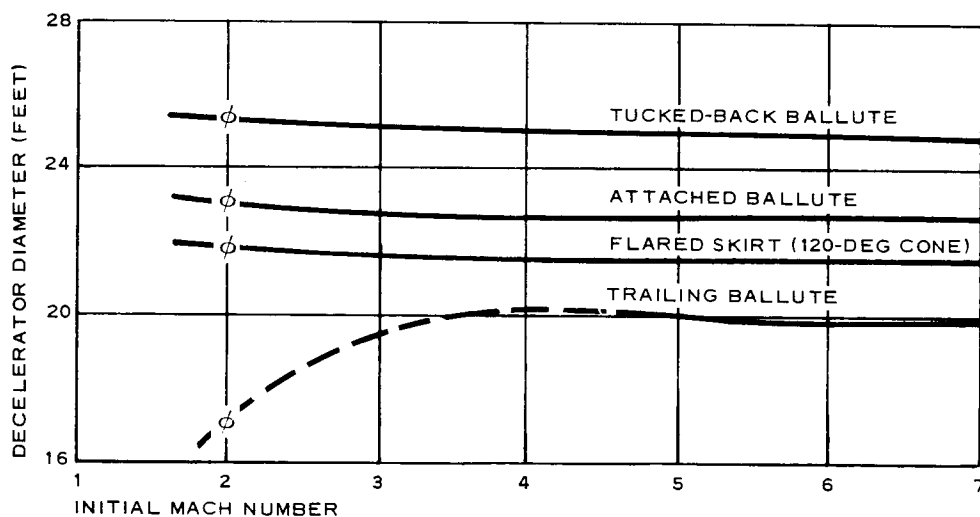


Figure 85 - Decelerator Diameter versus Deployment Mach Number for 16-Ft Diameter Entry Capsule, Trajectory 22 at 20,000 Ft

NOTES:                    $\phi$ = SELECTED MACH NUMBER          MASS (SLUGS): 84.5

   TRAJECTORY DESIGNATION: 22      A = NOMEX 600 F          $(C_D A)_V$ : 281.5 SQ FT

   ATMOSPHERIC PROFILE: VM7        B = DACRON 350 F

   TARGET MACH NUMBER: 1.0      ENTRY CAPSULE DATA:          $W_{EARTH}$ = 2720 LB

   TARGET ALTITUDE: 20,000 FT      DIAMETER: 16 FT

ATTACHED BALLUTE

TUCKED-BACK BALLUTE

TRAILING BALLUTE

FLARED SKIRT (60 DEG - 1/2 $\phi$)

DECELERATOR-TO-TOTAL WEIGHT (PERCENT)

INITIAL MACH NUMBER

INITIAL MACH NUMBER
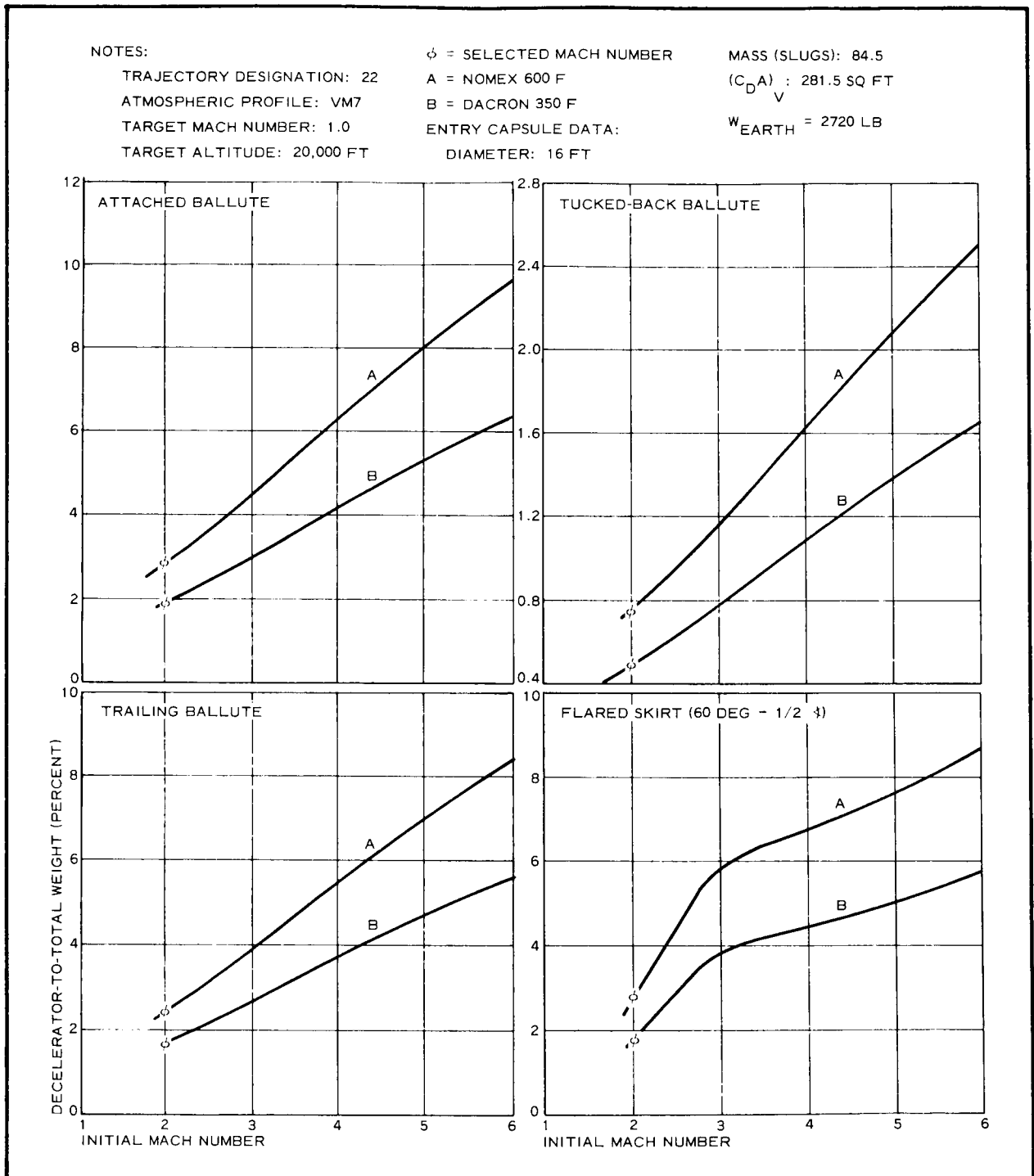
Figure 86 - Percent Decelerator to Total Weight versus Deployment Mach
Number for 16-Ft Entry Capsule, Trajectory 22 at 20,000 Ft

NOTES:

TRAJECTORY DESIGNATION: 22

ATMOSPHERIC PROFILE: VM7

TARGET MACH NUMBER: 1.0

TARGET ALTITUDE: 30,000 FT

$\phi$ = SELECTED MACH NUMBER

ENTRY CAPSULE DATA:

DIAMETER: 16 FT

MASS (SLUGS): 84.5

$\left( C_D A \right)_V$ : 281.5 SQ FT

$W_{EARTH}$: 2720 LB
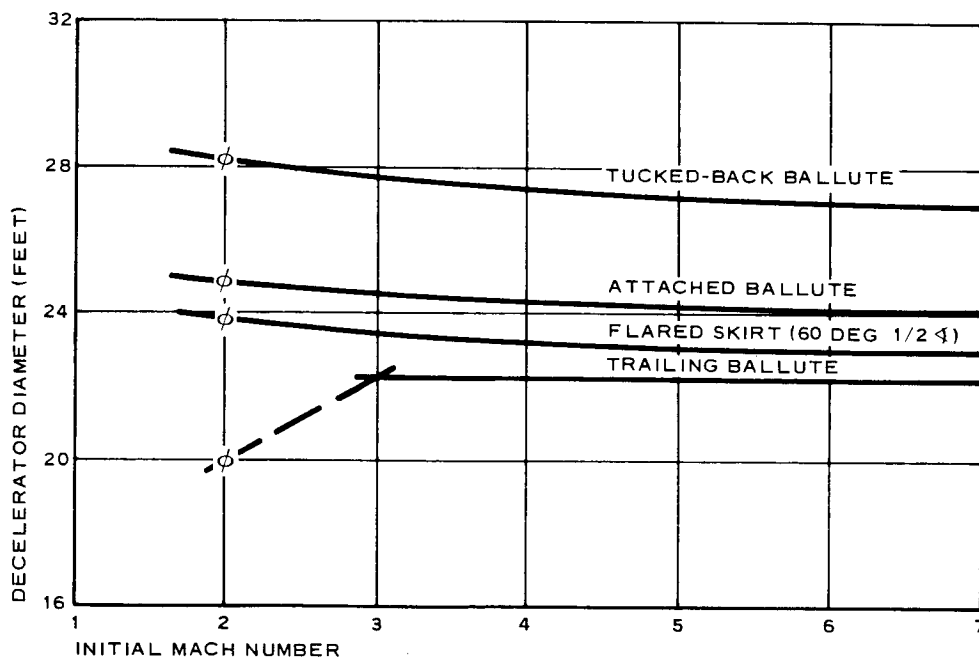


Figure 87 - Decelerator Diameter versus Deployment Mach Number for 16-Ft Diameter Entry Capsule, Trajectory 22 at 30,000 Ft

NOTES:

TRAJECTORY DESIGNATION: 22          ENTRY CAPSULE DATA:

ATMOSPHERIC PROFILE: VM7              DIAMETER: 16 FT

TARGET MACH NUMBER: 1.0              MASS (SLUGS): 84.5

TARGET ALTITUDE: 30,000 FT          $(C_DA)_V$ : 281.5 SQ FT

$\phi$ = SELECTED MACH NUMBER        $W_{EARTH}$ = 2720 LB

A = NOMEX 600 F

B = DACRON 350 F
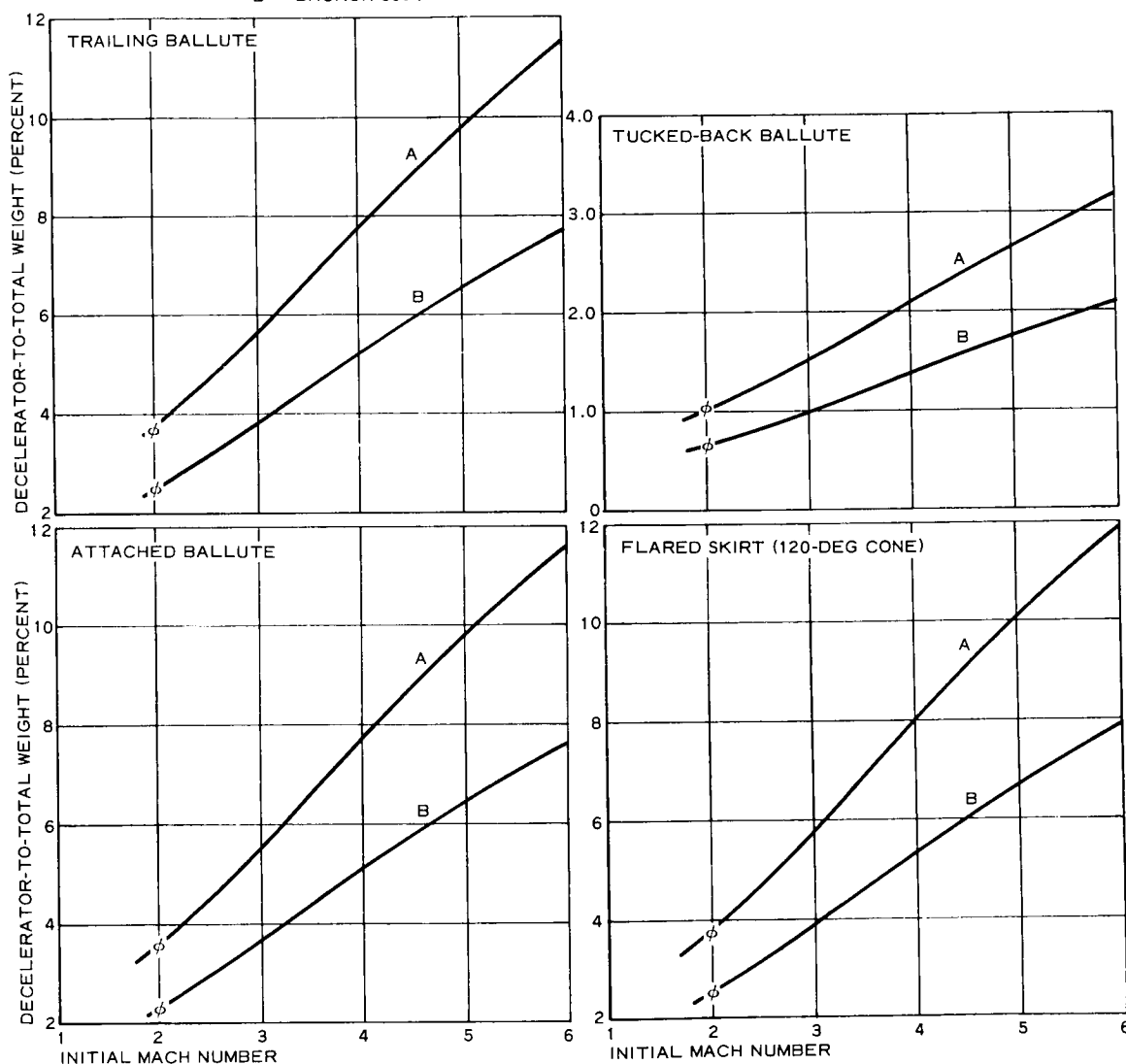


Figure 88 - Percent Decelerator to Total Weight versus Deployment Mach Number for 16-Ft Entry Capsule, Trajectory 22 at 30,000 Ft
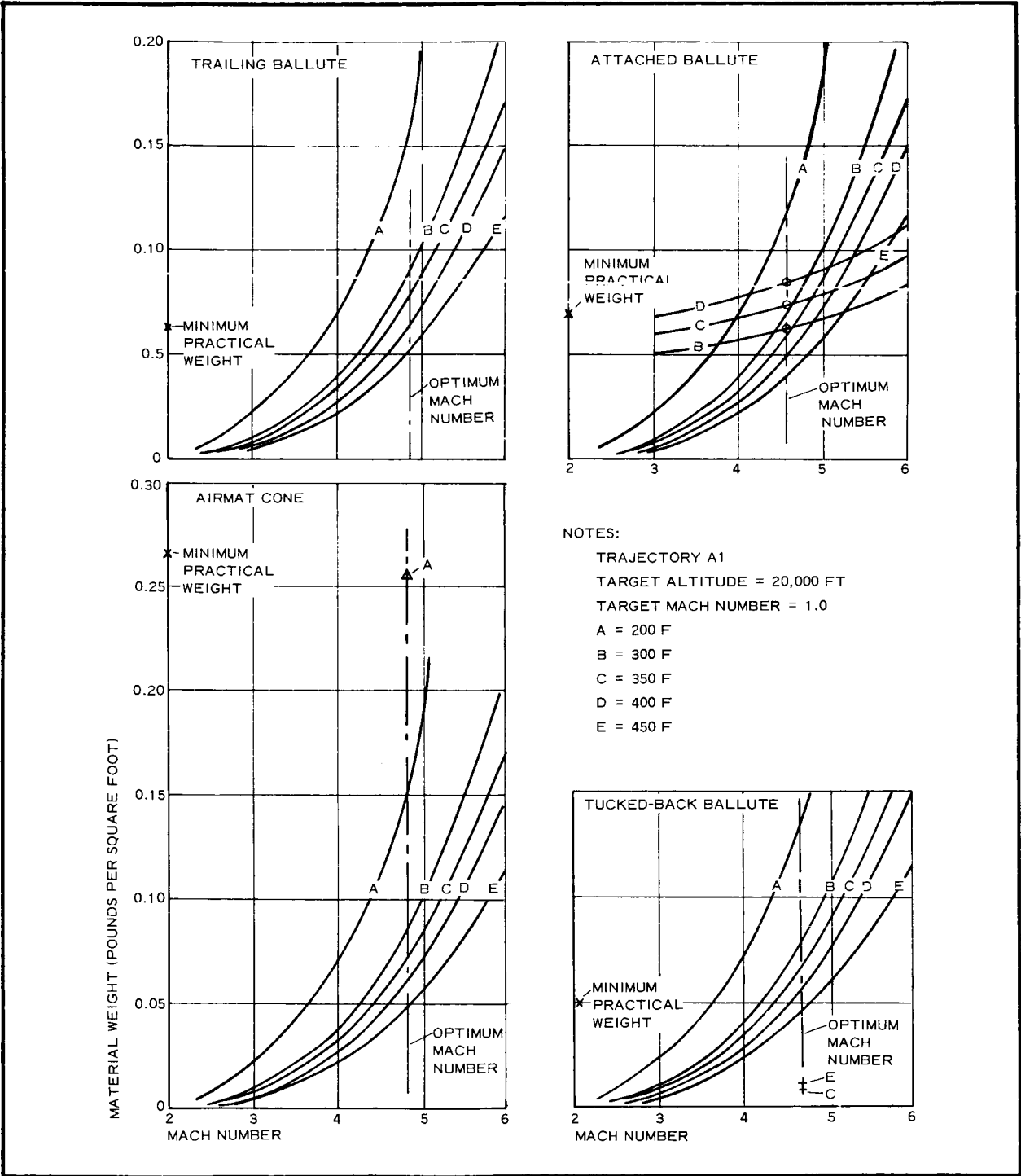
Figure 89 - Decelerator Envelope Unit Weight versus Deployment Mach
Numbers at Trajectory A1 (Dacron)
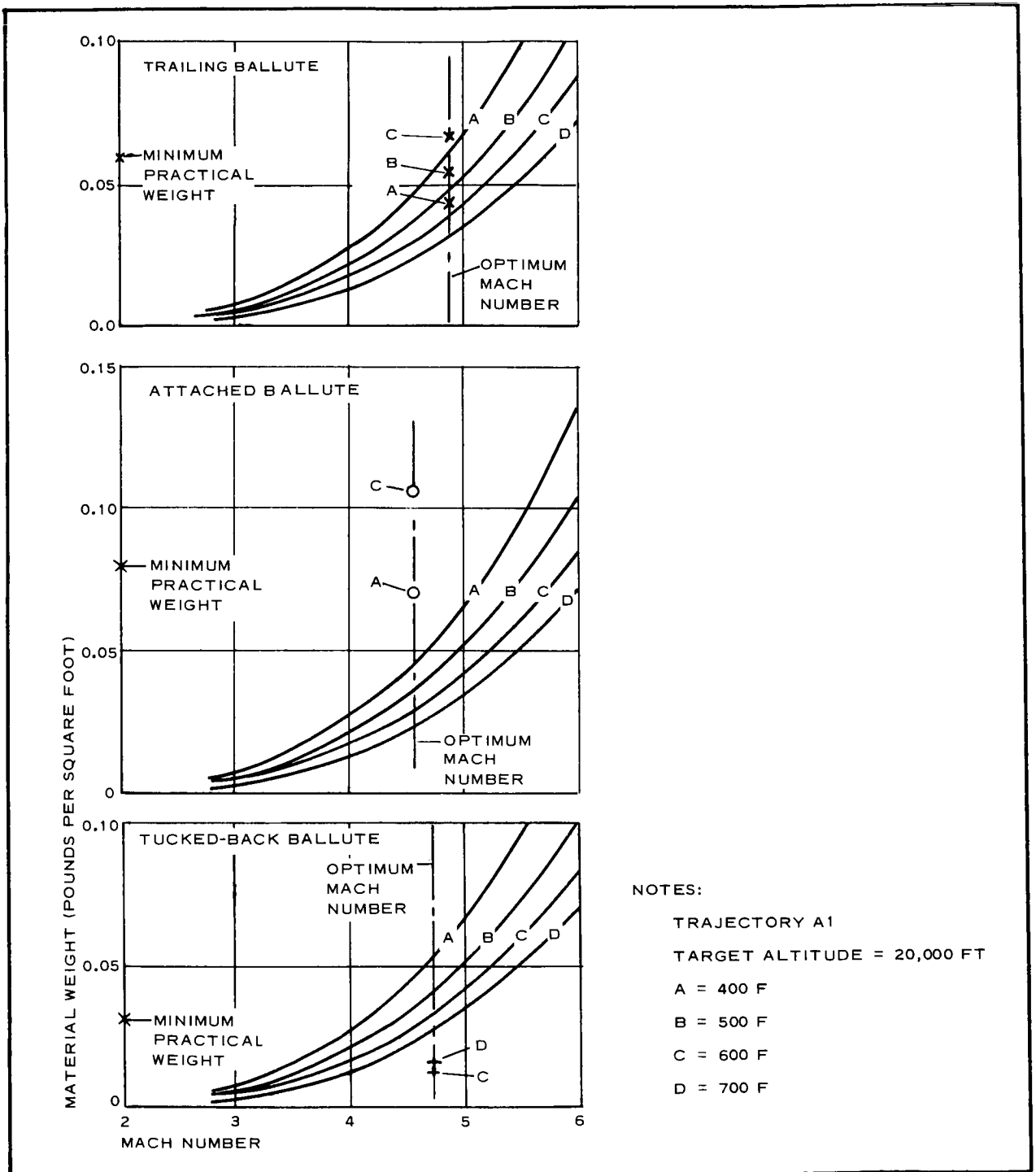
Figure 90 - Decelerator Envelope Unit Weight versus Deployment Mach
Number, Trajectory A1 (Nomex)
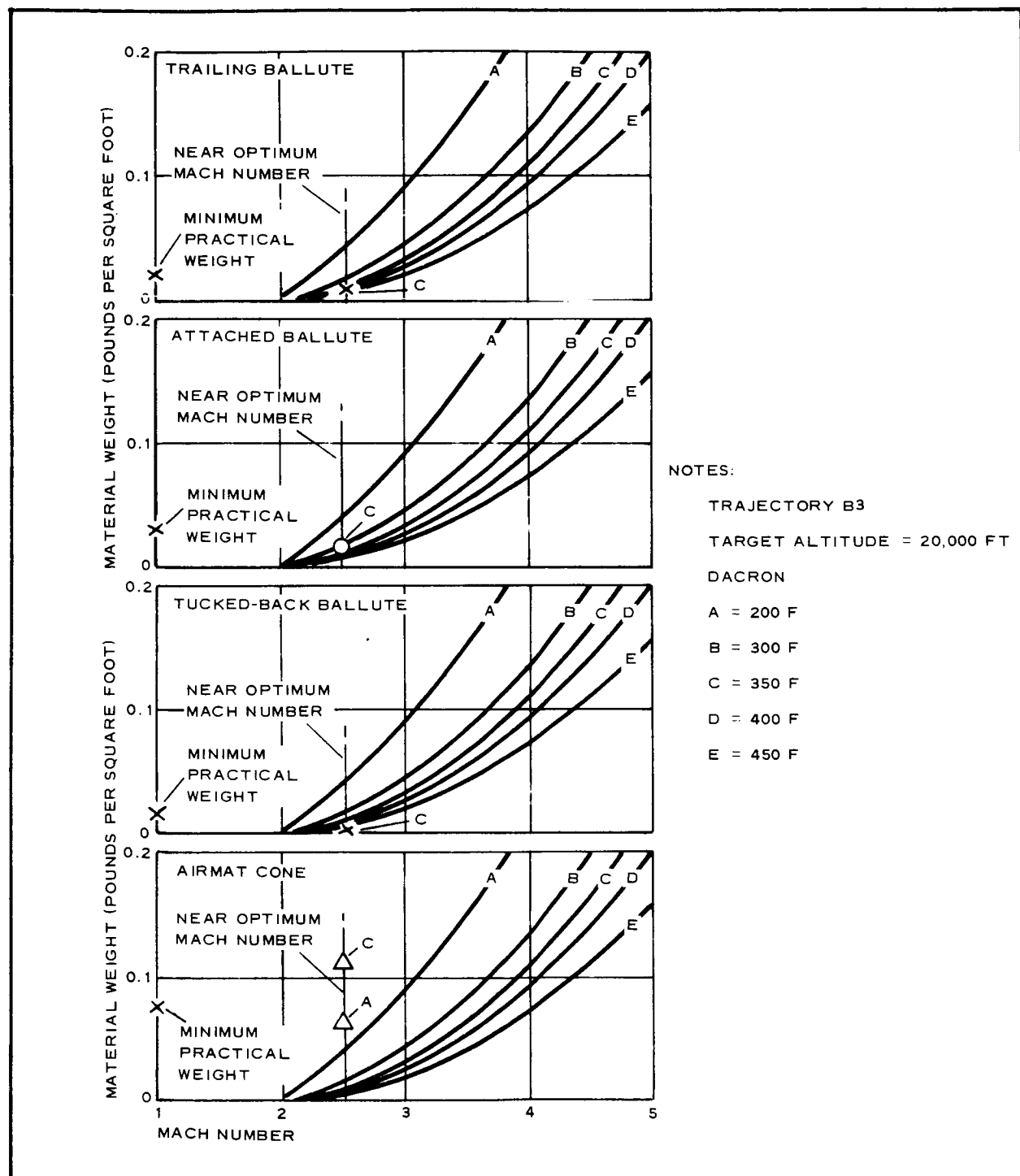
Figure 91 - Decelerator Envelope Unit Weight versus Deployment Mach
Number, Trajectory B3

NOTES:

TRAJECTORY 19

TARGET ALTITUDE = 20,000 FT

DACRON

A = 200 F

B = 300 F

C = 350 F

D = 400 F

E = 450 F
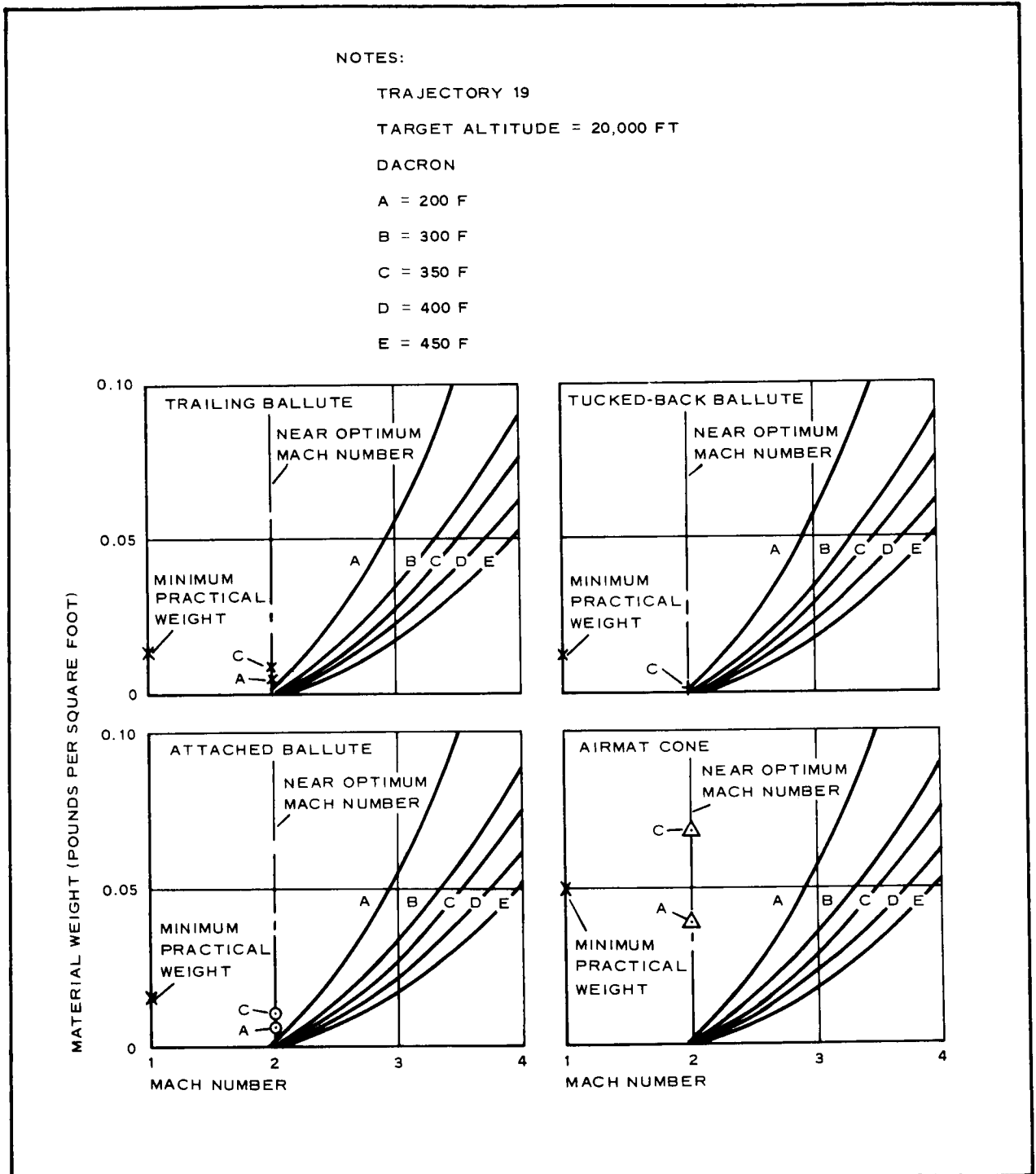
Figure 92 - Decelerator Envelope Unit Weight versus Deployment Mach
Number, Trajectory 19 at 20,000 Ft
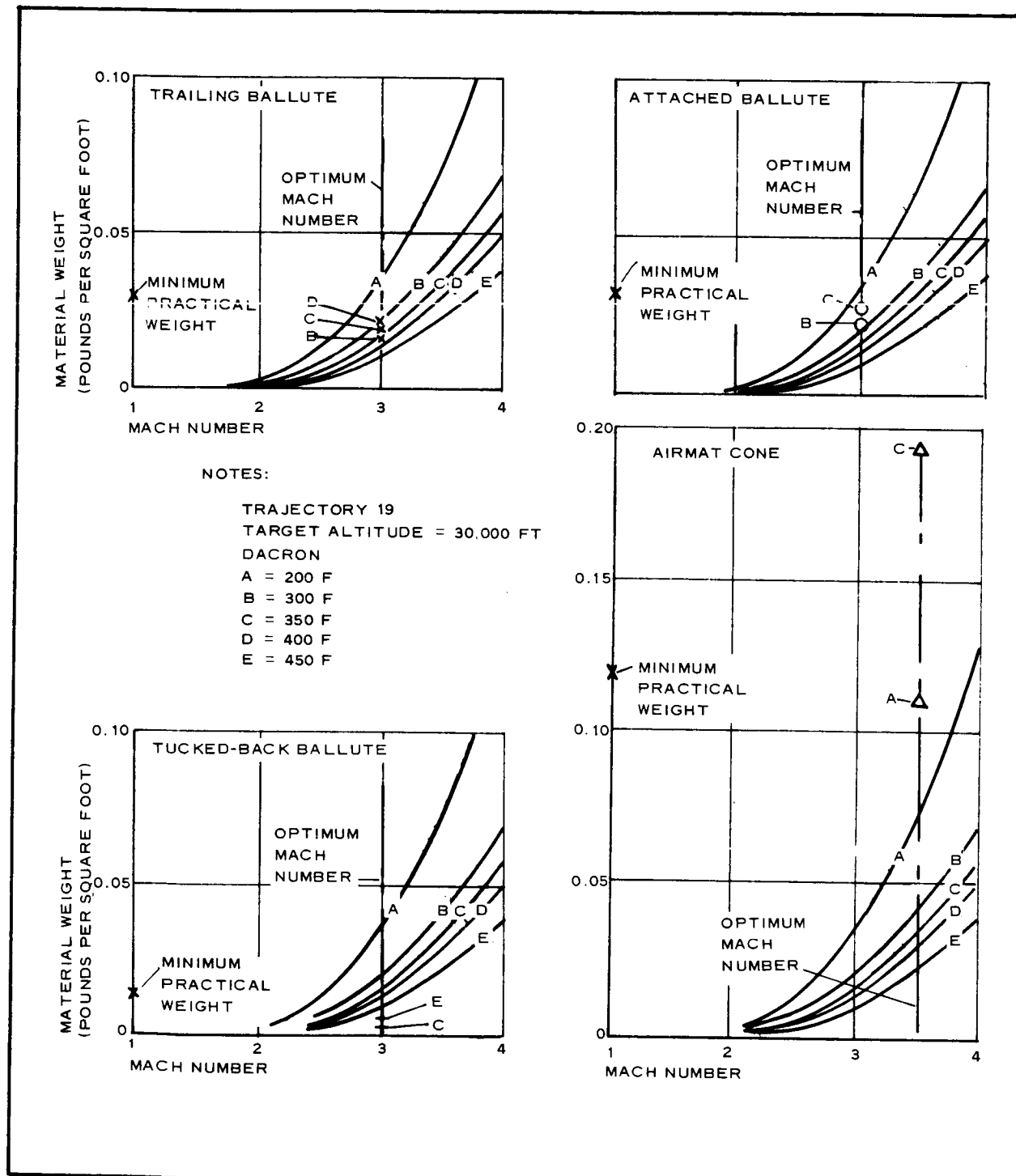
Figure 93 - Decelerator Envelope Unit Weight versus Initial Operating
Mach Number, Trajectory 19 at 30,000 Ft
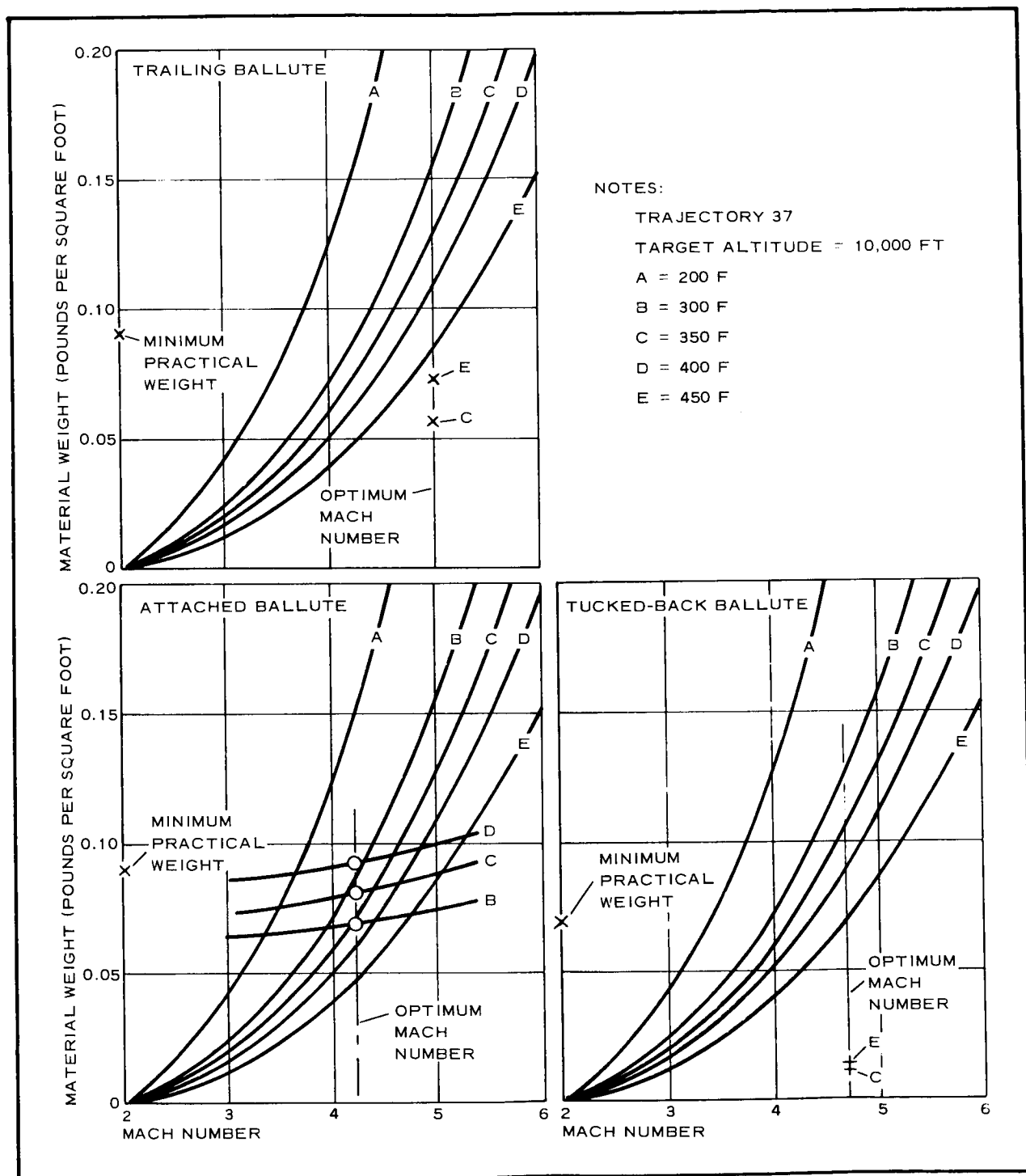
Figure 94 - Decelerator Envelope Unit Weight versus Deployment Mach
Number, Trajectory 37 (Dacron)
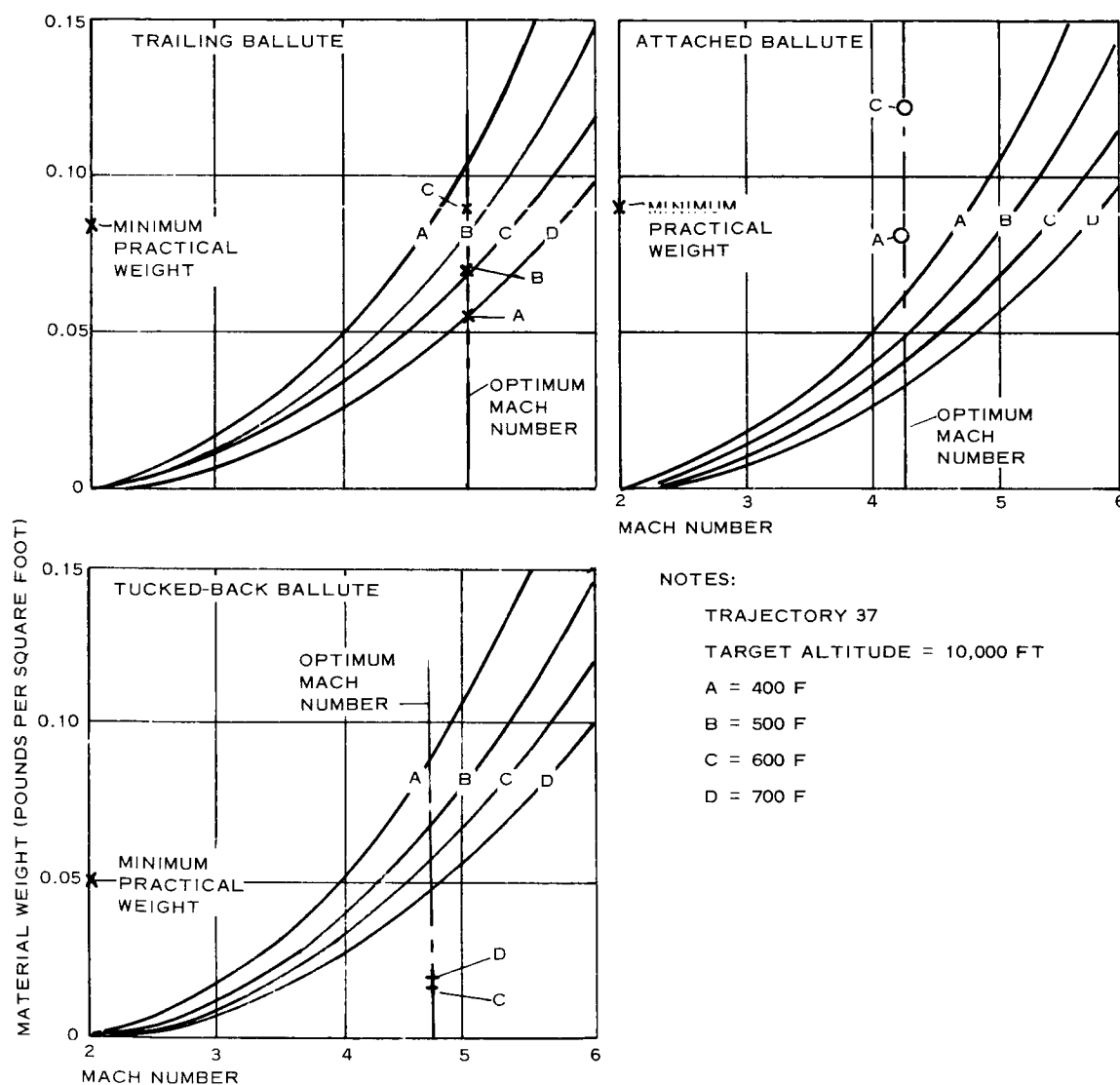
Figure 95 - Decelerator Unit Weight versus Deployment Mach Number, Trajectory 37 (Nomex)
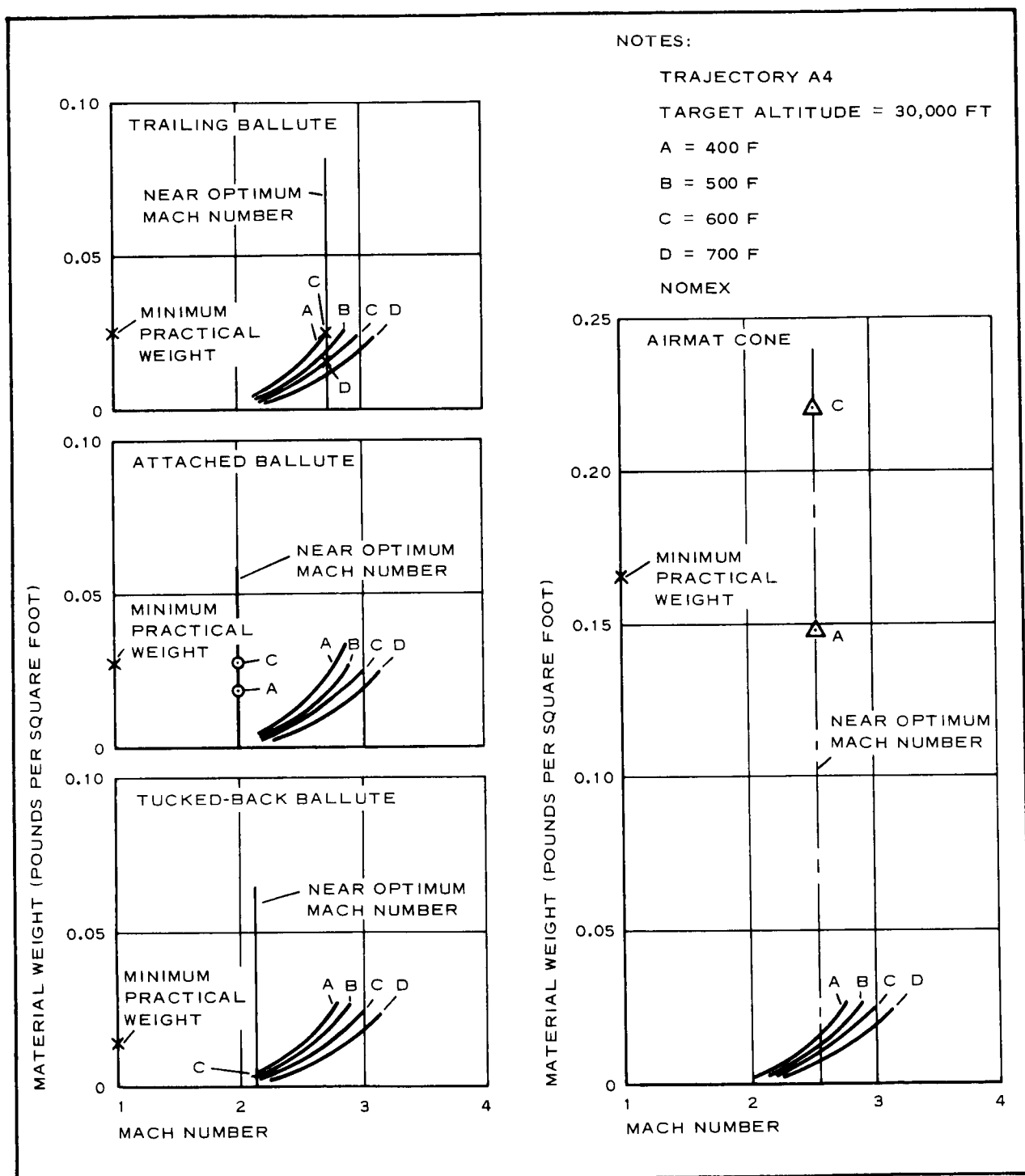
Figure 96 - Decelerator Envelope Unit Weight versus Deployment Mach
Number, Trajectory A4
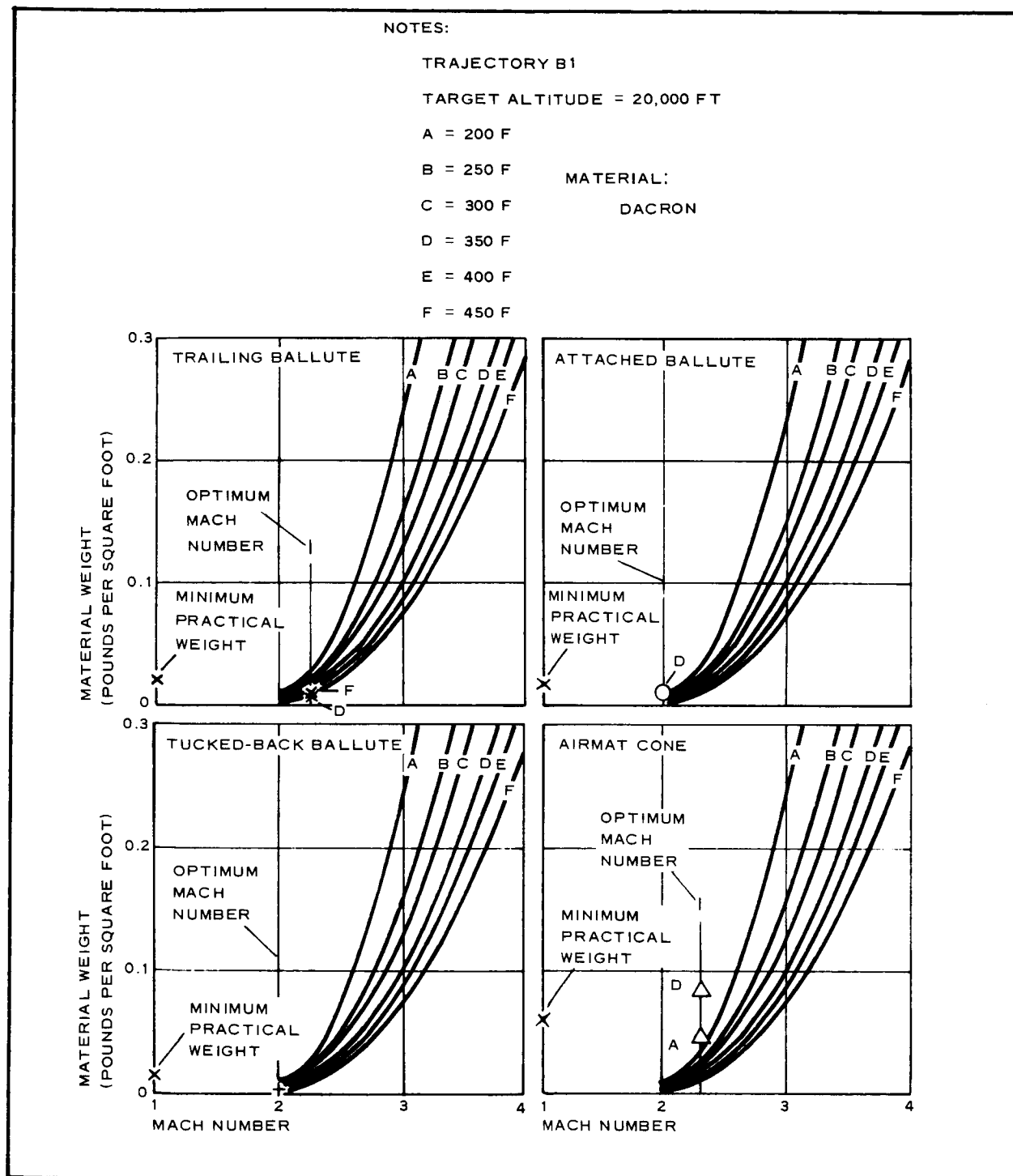
Figure 97 - Decelerator Envelope Unit Weight versus Deployment Mach
Number at 20,000 Ft

Figure 98 - Decelerator Envelope Unit Weight versus Deployment Mach Number, Trajectory B1 at 30,000 Ft
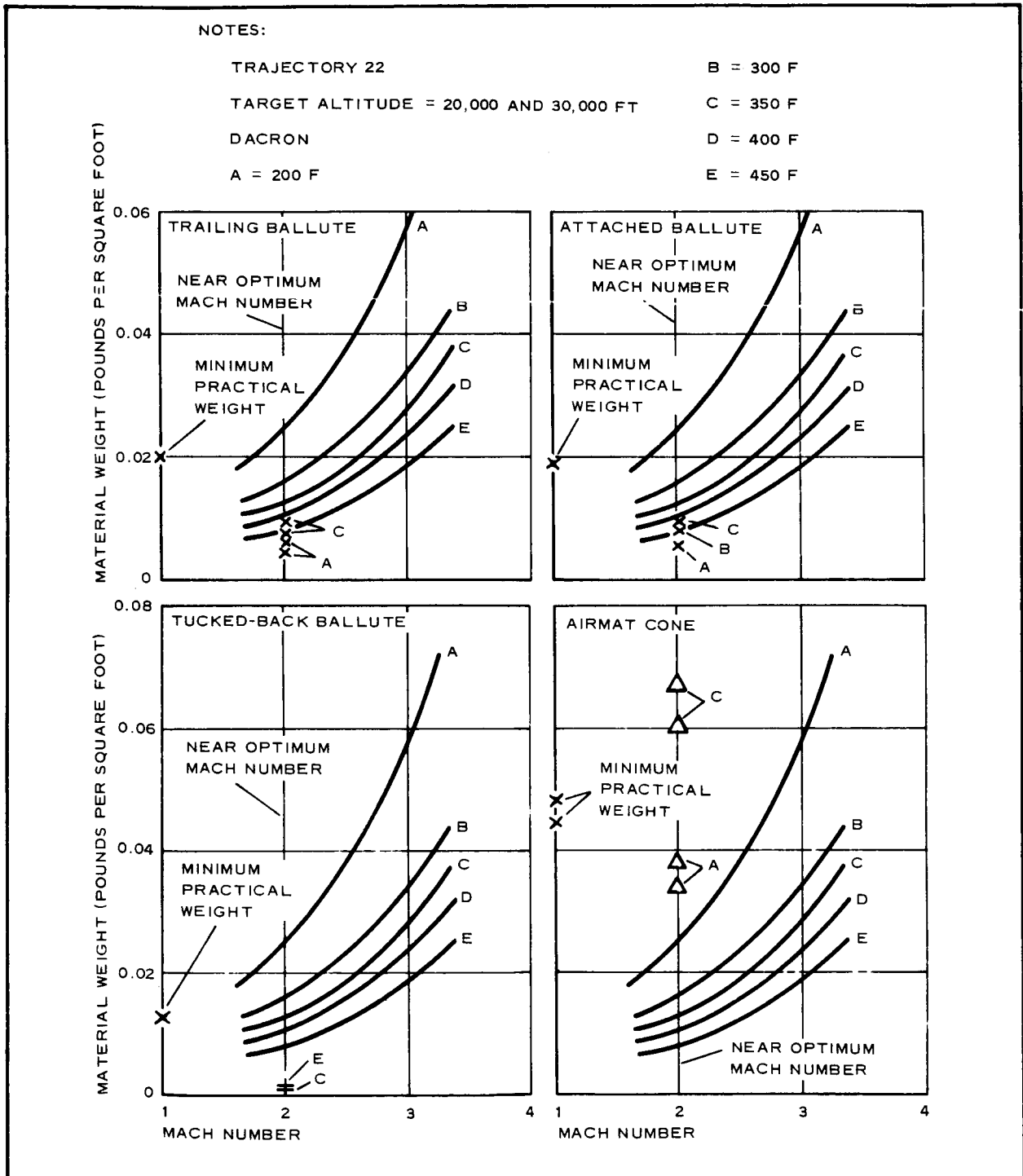
Figure 99 - Decelerator Envelope Unit Weight versus Deployment Mach
Number, Trajectory 22

# SECTION VII - AUXILIARY GAS-INFLATION MEANS FOR
# MARS ENTRY DECELERATORS

1. GENERAL

A gas-inflation system provides sufficient internal pressure to maintain a symmetrical rigid shape for an inflatable aerodynamic decelerator. Because decelerators can be applied in various stabilization and deceleration modes, each with peculiar thermal and structural considerations and a variety of altitude velocity regimes, inflation systems must be optimized separately.

Weight factors, prime considerations in determining acceptability of any design, are summarized in Figure 100 for more feasible methods of decelerator inflation. Systems considered were: (1) compressed gas in pressure vessels, (2) residual gas within the decelerator, (3) gas generation by burning fuel, (4) chemically generated gas, (5) auxiliary compressors, and (6) ram-air inflation.

2. COMPRESSED GAS IN PRESSURE VESSELS

Probably the most widely used method of delivering large volumes of gas at relatively low pressure is high-pressure release from relatively small-volume bottles made of steel, or fiberglass or bottles wound with wire.

The state-of-the-art in this area has progressed with development of metals and compression equipment with greater strength. Associated tubing, solenoids, and pyrotechnic valves have become off-the-shelf items. Inflation rates may be controlled easily with standard metering hardware.

Weight curves for steel-bottle and fiberglass-bottle systems in Figure 100 are based on catalog weights for commercially available bottles. Heat treated Ti-6A-4V alloy was selected and design conditions for the bottles
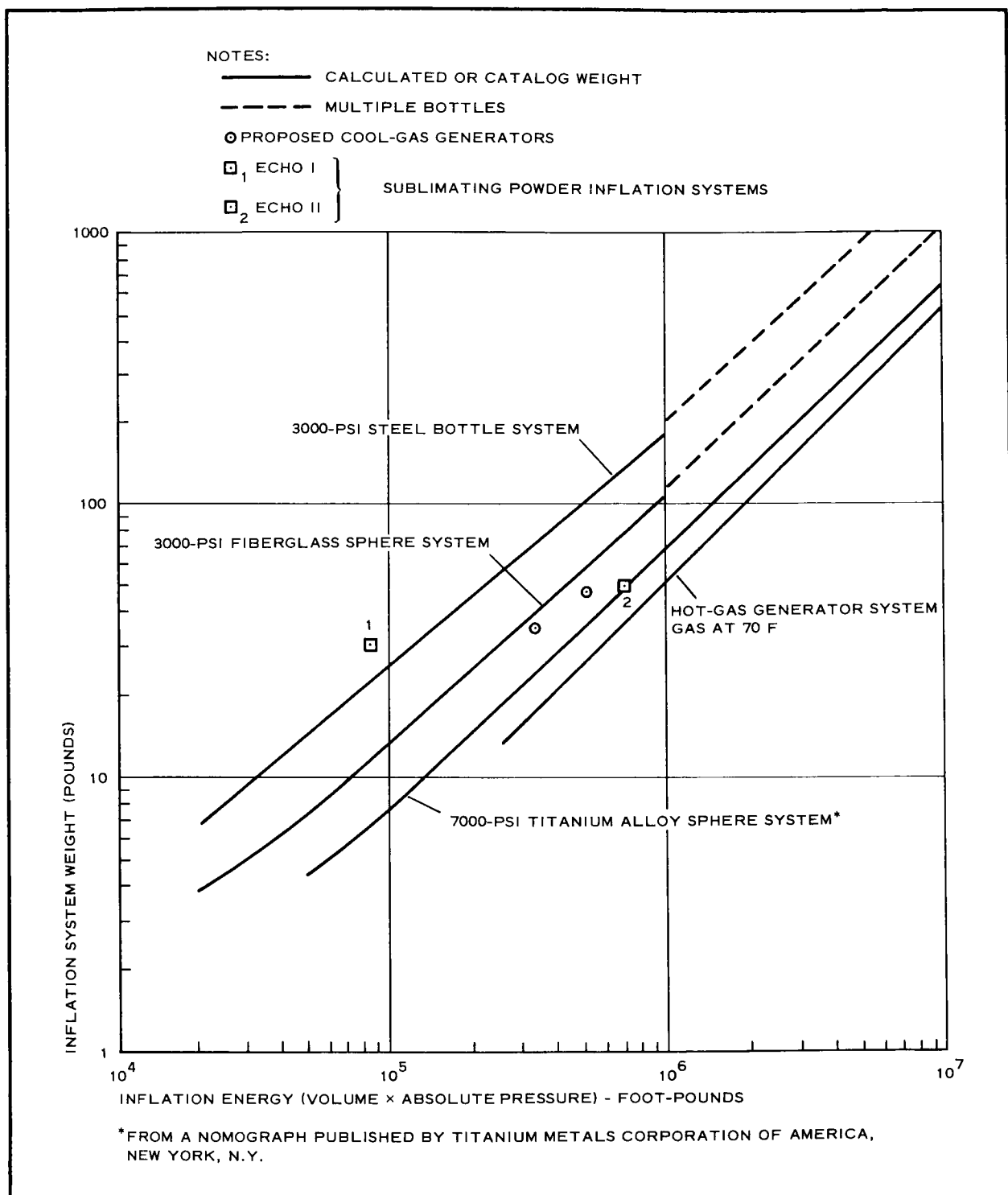
Figure 100 - Weights for Various Inflation Systems

were assumed to be 7000-psi pressure and 100,000-psi design stress. This stress level gives a safety factor of 1.6 at 70 F. At elevated temperatures corresponding to projected sterilization requirements for the entry capsule, the safety factor would be approximately 1.0.

Weight of each of the three gas-bottle systems included compressed air and the weight of the control valve. The control valve, assumed to be simple, was estimated to weigh from 1 to 3 lb over the range of bottle sizes.

3.  RESIDUAL GAS WITHIN THE DECELERATOR

The extremely low density of the Mars atmosphere suggests the feasibility of using gas trapped inside the packaged decelerator for inflation. If necessary to eliminate predeployment stresses on the decelerator, the packaging canister may be sealed hermetically creating a zero pressure differential across the fabric. This system ensures rapid inflation and good reliability and eliminates need for a pressure vessel and valves.

4.  GAS GENERATION BY BURNING SOLID FUEL

Generation of gas by burning "explosive" materials is reliable and being refined continually. Hot-gas and cool-gas generators are available in various sizes as off-the-shelf items.

Custom-made gas generators may be procured to fulfill most pressure volume requirements. These devices can be fired electrically or mechanically.

The gas generator curve in Figure 100 was calculated from the equation

$$Pv = wrT \tag{7}$$

where

$Pv$ = energy (lb-ft),

$w$ = propellant weight (lb),

T  =  gas temperature (F),  and

r  =  energy constant (55 ft per deg R - a good
average value for present-day ballistic
generators).

Total weights of gas generator systems were determined arbitrarily by
multiplying propellant weight by 1.5 to include weights of the case and
hardware.  The 1.5 factor, estimated from limited data on hot-gas gen-
erators, is based on an assumption that gas is generated hot and cooled
to 70 F inside the decelerator envelope.

Two points in Figure 100 show the weights of two cool-gas generator sys-
tems proposed by McCormick-Selph Associates of Hollister,  Calif. for
specific inflatable devices.

5.    CHEMICALLY GENERATED GAS

Applications to inflate decelerators at extremely low-density conditions
can use subliming solids or vaporizing liquids as the gas source.  Be-
cause this method can produce only very low pressures, its usefulness
is limited to initial inflation of ram-air devices in low-density-altitude
regimes.

Two examples of sublimating powder systems are shown in Figure 100.
The points were calculated from data for Echo I and Echo II balloon in-
flation systems (see Table VII).

### TABLE VII - ECHO I AND II INFLATION DATA

| System | Sublimating solid | Weight (lb) | Density (psf) | Pressure (mm Hg) | Volume (cu ft) |
|---|---|---|---|---|---|
| Echo I | Benzoic acid | 10 | 79 | | |
| | Antraquinone | 20 | 89 | | |
| | | 30 | 84 (average) | 0.06 | 525,000 |
| Echo II | Acetanide | 50 | 72 | 0.20 | 1,290,000 |

6.   AUXILIARY COMPRESSORS

A cursory feasibility examination of auxiliary compression equipment revealed several unfavorable characteristics. Compression of rarefied gases comparable to the Mars atmosphere imposes severe tolerance restrictions on the moving compressor parts. Time required is prohibitive for pumping large volumes of low-density gases into an inflatable device. Weights and volumes of airborne compressors compare unfavorably with other systems.

7.   RAM-AIR INFLATION

Ram-air inflation has been proven practical in subsonic, supersonic, and hypersonic regimes. Basically, the ram-air concept converts kinetic energy of the external dynamic pressure to internal static pressure by presenting orifices normal to the airstream. The inflation aperture can be located in almost any area exposed to the airstream. Internal recovery pressures one to four times greater than dynamic external pressures have been noted in the supersonic range when side inlets were used with the BALLUTE. This pressure boost capacity indicates that configurations requiring inflation pressures greater than the free-stream dynamic pressure also may be ram-air inflated.

In contrast to closed inflation systems, the ram-air method operates in a wide range of dynamic pressure conditions without over-pressurizing the inflatable device. This flexibility avoids the need for programming inflation gas, thus reducing the structural requirements of the fabric.

Inflation rate and g loads imposed on the payload can be controlled by the size of the inflation orifice.

8.   PACKAGE BULK FACTORS FOR THE VARIOUS INFLATION SYSTEMS

Package density, the ratio of weight to volume of the inflation system, was calculated for each method as follows:

| System | Density (pcf) |
|---|---|
| Hot-gas generator[a] | 60 |
| Cool-gas generator[b] | 95 |
| Fiberglass sphere, air at 3000 psi | 35 |
| Titanium sphere, air at 7000 psi | 59 |
| Steel bottle, air at 3000 psi | 63 |
| Sublimating powders | |
|     Echo I (benzoic acid and antraquinone) | 85 |
|     Echo II (acetanide) | 72 |

[a] Based on data of Frankford Arsenal, Philadelphia, Pa.

[b] Based on data of McCormick-Selph Associates, Hollister, Calif.

# SECTION VIII - TESTING REQUIREMENTS, TECHNIQUES, AND FACILITIES

1. GENERAL

   To evaluate accurately the performance of expandable decelerators, test-
   ing must be conducted under controlled or known conditions. These tests
   will determine important aeroballistic characteristics and structural in-
   tegrity of the decelerator design and its effectiveness in decelerating and
   controlling Mars entry capsule systems. These tests must include speci-
   fic areas of investigation described in Items 2 through 6 below.

2. STATIC AERODYNAMIC COEFFICIENTS AND AERODYNAMIC FLOW-
   FIELD EFFECTS

   Static force and moment data must be obtained for the Mars entry capsule
   as functions of angle of attack, yaw, and roll for variations of expandable
   decelerator drag area related to operating Mach number ranges. In addi-
   tion to measurements of static loads and moment, data must be obtained
   on the effects of shock-wave and effects of boundary-layer interaction,
   separation, and transition. These viscous flow phenomena will strongly
   influence the performance of entry capsule decelerator composites. Con-
   sequently, it is necessary in the experimental test program to control as
   closely as possible the correspondence and simulation of Mach number,
   Reynolds number, and flow conditions expected in the actual entry tra-
   jectory during decelerator operation.

   Static aerodynamic derivatives can be obtained in transonic and super-
   sonic wind-tunnel facilities with scale models supported by strain-gage
   instrumented sting balances. Aerodynamic flow-field effects can be de-
   termined with optical techniques such as Schlieren or shadowgraph photog-
   raphy or, perferably, with motion photography. Several configurations
   should be tested to study the effects of variations in decelerator geome-
   try and arrangements.

These data are necessary to ensure desirable static stability margins for the decelerator composite and to determine decelerator effectiveness during deployment.

3.   DECELERATOR LOADS AND LOAD DISTRIBUTION

Measurements to determine drag loading and load (pressure) distribution on decelerators must be made to estimate the induced moments and to determine the power requirements, bulk, and weight of the deployment and inflation mechanisms. In addition, the required structural integrity of the device itself will be determined by these measurements. A test program must determine unwanted rolling or yawing moments induced by flow asymmetries.

These data can be estimated from wind-tunnel tests by using a six-component sting balance and by providing known asymmetries.

4.   DYNAMIC STABILITY

For symmetrical decelerators the variations of pitching moment coefficients with pitching velocities and rates of change in angle of attack must be measured to evaluate dynamic stability of the vehicle at critical points on the entry trajectory. Generally, the values for these derivatives are most important when the negative rate of change in dynamic pressure is maximum following initial deployment and inflation of the decelerator device.

These derivatives can be measured by model tests in high-speed wind tunnels with an oscillating balance, with free-flight tests on a ballistic range or with photographic measurements in a wind tunnel.

5.   AERODYNAMIC HEATING

The decelerator must be tested in conditions of temperature and loading expected during entry into Mars atmosphere. These tests will determine the adequacy of materials and design integrity under these environmental conditions.

6.   TESTING METHODS

a.   General

Environmental conditions associated with Mars atmosphere entry
flight pose difficult problems in devising experimental study tech-
niques.  These arise principally from the extremely low anticipated
densities and pressures.

Figure 101 illustrates the range of Reynolds and Mach numbers that
possibly may be encountered in Mars atmosphere flight entry simu-
lated in the Earth's atmosphere.

Figure 102 shows the stagnation temperature levels encountered in
the Earth's atmosphere as functions of Mach number.  The solid por-
tion of the curve represents temperature values for the assumption
of equilibrium dissociation, and the dashed line is for no dissociation.
Dissociation is encountered as Mach number exceeds approximately
seven.  The temperature at which complete ionization of air occurs
is approximately 20,000 R and will not be reached for most contem-
plated recoverable Mars entry vehicles.

Much progress has been made in testing techniques and procedures
for study of the aerodynamics of high-speed flows.  However, at
present, no single laboratory technique is adequate to study all prob-
lems simultaneously.  Problems must be separated into parts that
can be studied with the laboratory techniques now available.  The
separate results then must be synthesized.

b.   Testing Facilities

Shock tubes, shock tunnels, and ballistic range facilities have been
significant contributions to gaining understanding of problems in the
flow regime of present interest.  These facilities include flow proces-
ses behind normal shock waves ahead of blunt bodies, boundary-layer
transition phenomena, skin friction and drag, and heat transfer as
inferred from skin friction.
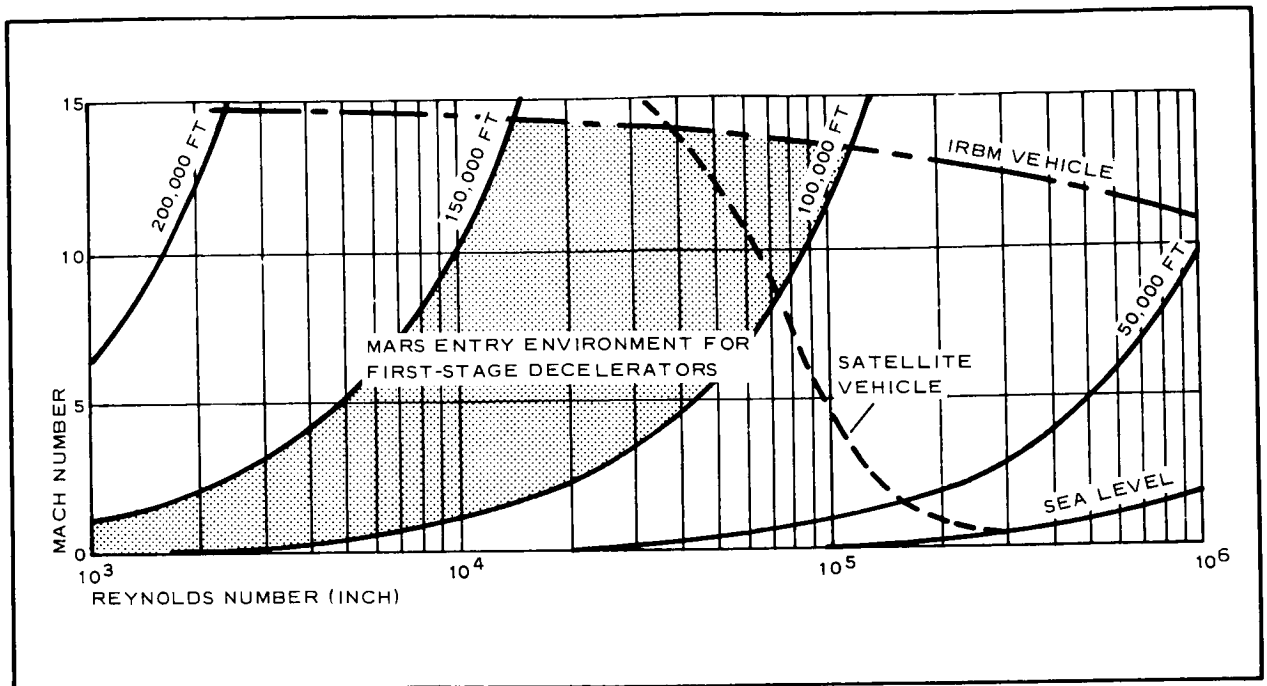
Figure 101 - Mach and Reynolds Numbers for Simulating Mars
Flight in the Earth's Atmosphere
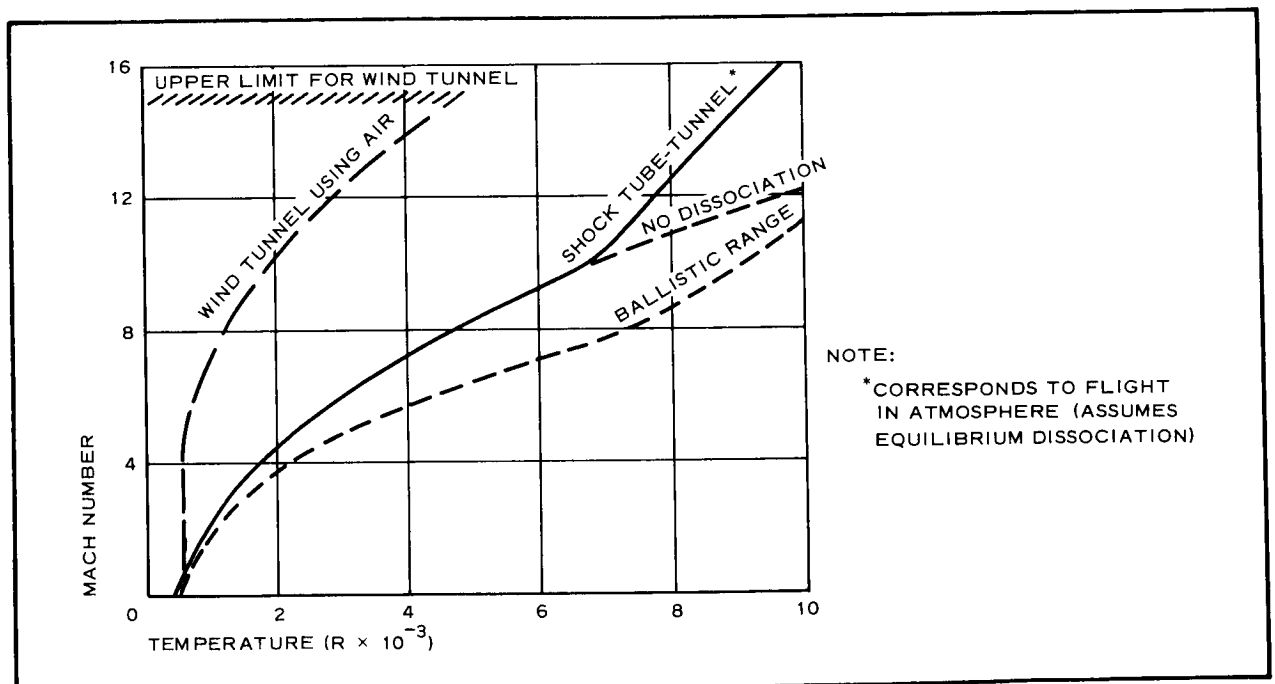


Figure 102 - Mach Number and Temperature for Atmospheric Flight and for
Various Laboratory Testing Methods

Figure 103 shows the range of Mach and Reynolds numbers and the
temperature level obtainable in wind tunnels using air superimposed
on the flight regions of Figure 101. The wind tunnel only allows cover-
age of a portion of the region of interest. In addition, at the low values
of Reynolds number required for simulation, accuracy of most wind-
tunnel data is limited.

Ballistic range facilities and free-flight model testing in wind tunnels
also can determine static aerodynamic force and moment coefficients
in addition to dynamic stability derivatives. Pressurized ballistic
ranges and wind tunnels make possible a reasonable flight Reynolds
number of the actual Mars entry capsule over the entire Mach number
range of interest for expandable decelerators shown in Figure 103.
Ballistic range facilities use light gas guns to propel the models to
test velocities and can accommodate models to 3 or 4 in. in diameter.
The required mass scaling to preserve dynamic similarity is some-
what difficult from the standpoint of size. However, the scaling is
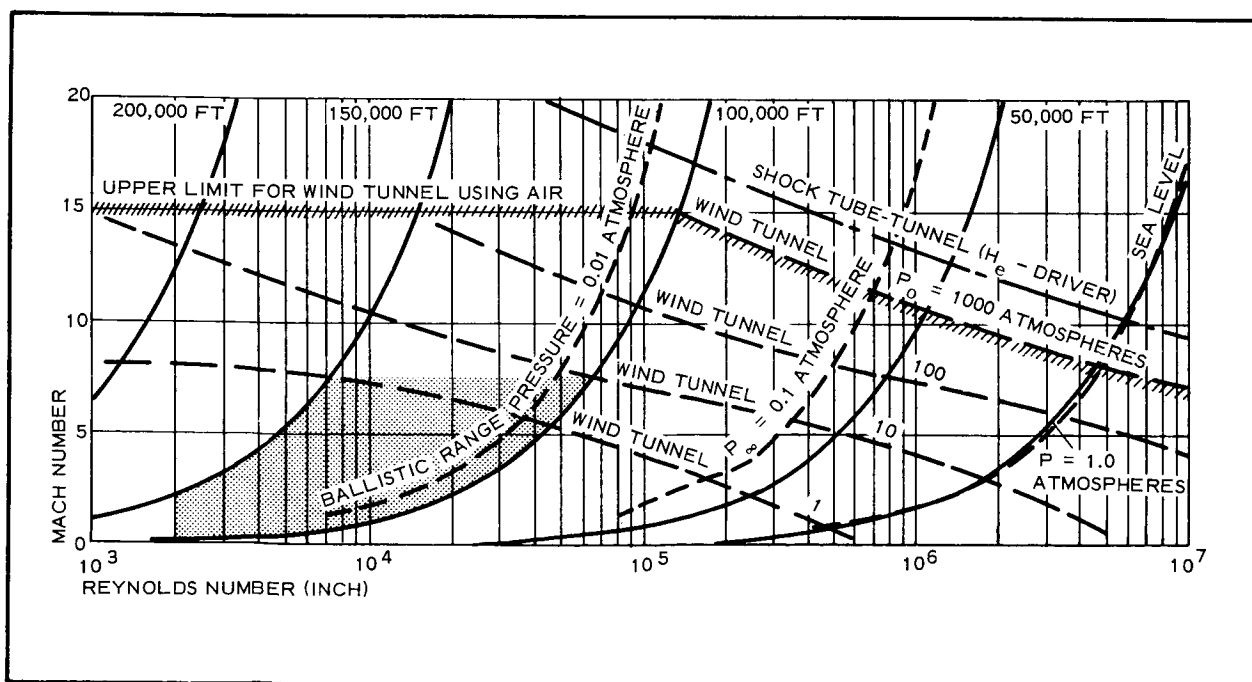


Figure 103 - Mach and Reynolds Numbers for Various
Laboratory Testing Methods

usually within the capabilities of the model propelling equipment for rotationally symmetric configurations of the type being considered. Thus, motion of the actual decelerator composite can be duplicated closely by the scaled model. In addition, all realistic interactions will be inherent in this testing method whereas unforeseen interactions of different effects may go undiscovered in wind-tunnel tests.

Ballistic range facilities also are turbulence-free and homogeneous, features desirable for the study of viscous effects such as boundary-layer transition and shock-wave boundary-layer interaction.

A disadvantage is that model sizes do not allow for extensive instrumentation. However, measurements can be made with photographs, shadowgraphs, and spectrographs of the model in flight and by observing the recovered model after test.

Free-flight model tests in wind-tunnels and ballistic ranges are limited by short testing times. Design study of expandable decelerators for Mars entry requires duplication of expected flight temperatures and rates of heating to determine how structures and structural materials react at high temperatures. Short running times of wind-tunnel and ballistic-range tests force the use of other tests for detailed investigations of such problems as aerothermo-elastic phenomena, cooling methods and surface chemistry with hot surfaces.

Heating rates experienced by Mars entry capsules could vary widely, depending on atmosphere, drag area, weight, velocity, altitude, and entry angle. For expandable decelerator applications considered in this study, the heat flux could range from 1 to 8 $Btu/ft^2/sec$.

Except for actual-free-flight testing, the power required to generate such heat fluxes over the large configurations required precludes simulation of aerodynamic heating on the full-scale body of the entry capsule.

c.   Heating Techniques

(1)   General

Techniques developed for research in the fields of decelerator materials and structures at temperatures that correspond to the anticipated Mars entry environment include: (1) electrical resistance heating, (2) electric blankets, (3) radio frequency heating, (4) hot body radiation, (5) wind-tunnel testing, (6) heating with hot gas jets, and (7) free flight testing.

These techniques are described in Items (2) through (8) below.

(2)   Electrical Resistance Heating

Electrical resistance heating, with current passed directly through the conducting part, is confined generally to testing of small-area specimens with uniform cross section; otherwise, the current requirements become excessive. Theoretically, the rate of temperature rise obtainable is almost limitless, depending only on the internal resistance of the material and the voltage applied.

(3)   Electric Blankets

An electric blanket relies on conduction for heating. Flexible and adaptable to irregular shapes, its use is confined to applications requiring slow heating, although large areas may be covered. Practical heating rates with this technique range from 0.1 to 4.0 $Btu/ft^2/sec$.

(4)   Radio Frequency Heating

Radio-frequency induction heating lends itself to rapid heating of reasonable size structural components of uniform material. This method closely approximates heating because the outer surface is heated. A disadvantage of this technique is that dissimilar materials will heat at different rates, causing problems in mounting test specimens. If rapid heating is desired, power requirements limit this method to testing of small sizes.

(5)  Hot Body Radiation

Several methods using hot-body radiation have been devised to
simulate aerodynamic heating.  One method uses a large elec-
trically heated graphite plate.  Electrical current through the
plate generates the heat by self-resistance.  A refractory oven
is used.  When the plate is withdrawn from the oven, it faces and
heats the test specimen by radiation.  Relatively large specimens
can be heated by this method with rapid temperature rises pos-
sible.  Longer heating periods are used, however, to reduce
power requirements.  Test specimens can be heated on both
sides of the plate simultaneously.  A similar, more flexible
method uses graphite rods heated to incandescence by a large
electrical current passed through them.

Another similar device is a tubular quartz lamp with tungsten
filaments.  Filament temperatures to 3500 F can be attained
with acceptable life.  Such lamps are satisfactory in terms of
flexibility and control.  By control of filament current, the heat
flux can be controlled and programmed.

(6)  Wind Tunnel Testing

Some research on structures has been done in wind tunnels using
air; however, this testing does not lend itself well to existing fa-
cilities, primiarily because structural failure could cause severe
wind-tunnel damage and delay the use of an expensive installation.
High-speed wind tunnel facilities have been developed for struc-
tural testing that incorporate some heating techniques discussed
in this section.

(7)  Heating with Hot Gas Jets

Hot gas jets for simulation of aerodynamic heating can heat the
material and simulate the effects of oxidation and erosion.  Oxi-
dation and erosion simulation require that the oxygen content be
proportional to the atmosphere and that the gas move at high

velocity. Among these devices are ram-jet configurations sup-
plied with air from a preheated auxiliary air supply. The sup-
ply can be augmented with pure oxygen to maintain atmospheric
oxygen content at the nozzle exit. Tests at temperatures to
2000 F at M = 2 can be obtained. By injecting and burning fuel
such as ethylene gas, jet stagnation temperatures correspond-
ing to M $\approx$ 6 can be obtained.

Similar devices, modified rocket motors burning kerosene with
hydrogen peroxide or red fuming nitric acid with anhydrous am-
monia, have been useful for preliminary materials evaluation
and for research on insulation and cooling schemes. Although
the above methods provide simplicity and utility, the gas streams
are not real air. Work has been conducted at the NASA Langley
Laboratories to provide an uncontaminated air jet with stagna-
tion temperatures of approximately 4000 F. This device con-
sists of heating ceramic pebbles in a chamber. When the desired
temperature is reached, the heater is turned off and uncontami-
nated air pumped into the chamber and through a supersonic
water-cooled nozzle into a test chamber.

(8)  Free-Flight Testing

Free-flight testing is the one feasible way of attaining accurate
heat flux associated with flight at higher Mach numbers. This
technique would use high-performance rocket units in progres-
sive stages to which the test vehicle with decelerator is attached.
By varying trajectories or flight paths of these rocket-powered
test vehicles, a wide variety of test conditions can be attained.
Trajectories are varied by changing the number of stages and
the time-delays between stages. Firing all stages during the
ascending phase produces a straightaway trajectory. This tra-
jectory yields information on high-speed behavior at high alti-
tudes. An over-the-top trajectory is achieved by firing the latter

stages after the test vehicle has started to descend. This flight path yields information at lower altitudes and can simulate closely a Mars entry trajectory. —

This method of testing, although relatively expensive and requiring extensive instrumentation, has been successful. Section XI contains a comprehensive description of the free-flight test techniques.

Table VIII summarizes various experimental techniques that may be employed for the study and evaluation of expandable decelerator devices for Mars atmospheric entry.

TABLE VIII - EXPERIMENTAL TECHNIQUES FOR STUD

| Experimental technique | Suitable areas of study | |
|---|---|---|
| Rocket-propelled models | Heat-transfer and skin friction; boundary-layer transition; cooling methods; characteristics of materials and structures at high temperatures and heating rates; and vehicle lift, drag, and stability including dynamic behavior. | Cor num |
| Ballistic ranges and free-flight, wind-tunnels | Skin friction and drag; boundary-layer transition; flow characteristics, including shock-wave, boundary-layer interaction; and heat-transfer and dynamic flight behavior (requires improved techniques and instrumentation for accurate measurements). | Cor num |
| Shock tubes and shock tube wind tunnels | Properties of air at high temperatures; relaxation effects; heat-transfer and boundary-layer phenomena for some conditions; and flow configurations, pressure distribution. | Cor num or num ber |
| Intermittent tunnels of "hotshot" type | Similar to those suitable for shock-tube study. | Cor num abo |
| "Hot" supersonic tunnels with long testing times and heating by:<br>1. Combustion of fuels<br>2. Air compression<br>3. Ceramic storage heaters<br>4. Combinations of (2) and (3)<br>5. Electric arcs. | Heat transfer to moderately high Mach numbers; study of cooling methods; behavior and development of materials and structures, including aerothermoelastic problems; and aerodynamic problems to Mach numbers approaching 15. | Cor num aro pra |
| "Cold" supersonic tunnels | Aerodynamics of perfect gas flows; model configuration studies; exploratory aerodynamic research to high Mach numbers. | Ma usi |

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Y OF EXPANDABLE DECELERATORS

| Simulation capabilities | Limitations |
| --- | --- |
| rect flight enthalpy, Mach iber, and Reynolds number. | Relatively high cost, inability to observe phenomena at close range, complexity of over-all system operation degrading reliability of data acquisition |
| rect flight enthalpy, Mach iber, and Reynolds number. | Model size is small and short testing time makes difficult the accurate measurements of more than a few relatively simple quantities |
| rect enthalpy and Reynolds iber with low Mach number correct enthalpy and Mach iber with low Reynolds number. | Millisecond testing times |
| rect enthalpy and Reynolds ibers to Mach numbers of ut 15. | Short testing times |
| rect enthalpy and Reynolds iber to Mach numbers of und 7. Continuous flow for ctical purposes. | Short testing time |
| ch numbers to about 9 or 10 ig air. | Incorrect enthalpies |

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## SECTION IX - FREE-FLIGHT TEST SIMULATION

## 1. GENERAL

This section discusses an earth atmosphere, free-flight test technique
for acquisition of data to evaluate deployment, inflation, performance,
and structural integrity of expandable terminal decelerators for a Mars
entry capsule. Concepts of this system are dictated primarily by relia-
bility, performance, and economic factors. The system will require re-
duced range and ground-support facilities compared with those used for
long-range ballistic missile or satellite systems. Complexity of program
operations is minimized and reliability is increased.

The functional performance evaluation of expandable decelerators for a
Mars entry capsule requires that consideration be given to free-flight
simulation in the Earth's atmosphere of all possible combinations of
loads and stresses anticipated for a range of nominal entry trajectories
to the Mars surface.

Experimental techniques available for controlled laboratory and captive
testing are limited in capability of simulating simultaneously the range
of environmental effects that could be encountered. Consequently, the
requirement for approximately full-scale, free-flight, proof testing un-
der conditions approaching environments anticipated during passage
through the Mars atmosphere.

Function and performance can be evaluated feasibly for expandable decel-
erators with existing range and ground-support requirements, for exam-
ple, those available at Wallops Island, Va., White Sands Missile Range,
N. M., or Kennedy Space Center.

A proven approach consists of launching a rocket-powered vehicle con-
trolled for near-vertical ascent ($\gamma_o \approx 80$ deg). Inherent aerodynamic

stability would be provided to control the launch vehicle and some spin would minimize dispersion caused by errors of thrust alignment, for example, lateral offset and unknown wind errors. The test vehicle is propelled to attain desired velocity and altitude in ascent or descent.

During descent tests, impact-point dispersion can be reduced by firing the last stage during re-entry. This firing will achieve necessary test velocity and altitude without the need for extreme summit altitudes.

Various government-sponsored programs have shown that this technique makes possible a wide range of velocity-altitude test combinations. Magnitude and time of impact pressures, deceleration loads, and heating are disadvantages. These will be higher than those for normally shallower entry angles anticipated for Mars entry unless more precise control of the vehicle flight path is provided. However, the aerodynamic effects anticipated for currently projected Mars entry flight paths for the most part can be simulated realistically by this technique.

2.  DESCRIPTION OF LAUNCH VEHICLE SYSTEM

    a.  General

        Selection of the propulsion units and staging for a rocket-boosted decelerator test vehicle must be based on analysis of such considerations as performance, range safety, aerodynamics, reliability, construction, and thermal effects. In determining motor staging, the ability to attain the desired test velocity and altitude is a primary consideration. Solid-propellant rocket motors have been developed to support rocket-propelled free-flight tests. Combinations of these motors can produce desired test velocity and altitude.

    b.  Booster Selection

        Booster sizes should be chosen for near-optimum mass ratios for the stages. Selection of the initial stage is influenced by thermal and inertial loading. To minimize aerodynamic heating and g load effects

during boost, a relatively long burning time for the initial stage is
desirable. Related also to booster selection are lead time required
for delivery and associated standardized hardware such as interstage
connections, stabilizing fin assemblies, ground service equipment,
launchers, and erectors. The current Aerodynamic Deployable De-
celerator Performance Evaluation Program (ADDPEP) program con-
ducted under Contract AF33(615)-1513 by Goodyear Aerospace has
integrated these selection considerations and has established and
proved in 16 rocket-boosted free-flight deployable decelerator tests
the reliability and performance of the selection method. For free-
flight entry simulation testing of the entry capsule, required summit
altitudes of 50 to 100 mi can be attained at launch angles near 80 deg
with a typical combination of XM33-Recruit and Lance boosters in
sequential stages. Firing the final stage may be desirable during
descent in an over-the-top trajectory to reduce summit altitudes and
impact-point dispersions. Whether firing all stages in ascent or re-
serving the last for re-entry, motor staging energy needs compara-
tively are almost equal.

3. LAUNCH VEHICLE FLIGHT CHARACTERISTICS

To establish adequately the performance of expandable terminal decel-
erators for the entry capsule, requirements must be determined for static
and dynamic stability of the launch vehicle. Required stabilizing surfaces
must be established for stability at the Mach range applicable at each
stage. Dispersion studies should determine effects of perturbations from
such influences as thrust malalignments, wind measurement or predic-
tion errors, launching inaccuracies.

Two conflicting requirements relate to aerodynamic stability of the launch
vehicle. High stability is desirable to minimize dispersion caused by
malalignments. Conversely, minimum stability will achieve minimum
dispersion from lateral winds. For an unguided vehicle, changing launcher
position for known winds can compensate to a great extent for in-flight

wind effects.  Changing launcher position restricts wind dispersion to errors in measurement or changes in the wind.

Center of pressure for the typical booster (without fins) is usually at an unstable location.  Thus, additional stabilization is needed for moderate aerodynamic stability; too much, however will cause the vehicle to become extremely sensitive to disturbance inputs.  Initial static stability of 5 to 6 percent of total vehicle length usually is adopted for a preliminary design.  A cruciform fin arrangement secures this margin.  Delta-shaped stabilizing fins might be selected for structurally efficient load-carrying capability.  The static margin desired, combined with increased fin effectiveness caused by fin-body interference, indicates that required fin sizes do not cause excessive performance reduction from aerodynamic drag.  First-stage static stability improves as fuel is consumed because the forward center of gravity shifts.  With increasing supersonic Mach numbers, the moment arm of a delta-shaped configuration generally will increase at a faster rate than the fin effectiveness decreases.  The moment arm is that distance from fin center of pressure to vehicle center of gravity.  Thus, the static margin of stability is usually least for a short interval just after launching.

Subsequent stages are designed for substantially greater static stability than the first stage because lateral winds are less forceful as vehicle velocity increases.  Flared skirts or fins at the base of the succeeding stage can provide a stability margin of approximately 15 percent of body length.  The stability margin for these stages also will increase during burning.  However, aerodynamic surfaces must be designed for stability despite possible reduction of body flare or fin effectiveness as higher supersonic Mach numbers are attained.

4.   DISPERSION CONTROLS

Dispersion of the decelerator test vehicle from its nominal trajectory can be caused by winds, aerodynamic asymmetries, thrust malalignments, variations in total rocket motor impulse, and reduced

aerodynamic damping caused by decreasing air density during ascent. Because of the relatively steep initial launch angle normally used in rocket-launched decelerator test programs to minimize range and tracking requirements, peak altitude will not be greatly affected by small dispersive influences. However, the steep launch attitude will increase range dispersion.

Dispersion caused by malalignments, asymmetries, and winds is greatly reduced when spin is used in staging. Low-altitude winds will have the greatest effect on the vehicle trajectory because vehicle velocity is low. Usually, greatest stability demands occur shortly after launch.

Rotating and tipping the launcher to account for changes in launch angle caused by the wind can compensate for wind dispersion. The actual wind dispersion then is associated with errors in wind measurement or wind changes between measurement and launch.

For test missions to be conducted, ranges of approximately 500 naut mi can be anticipated. Although severe dispersions may be considered, missions of this type could be conducted at Wallops Island or at KSC.

Experience with ADDPEP has shown that dispersion from thrust malalignment may be reduced by employing larger fins to increase stability. This approach, however, increases vehicle wind sensitivity. Higher spin rates also may be employed to reduce dispersion from thrust malalignment. At a certain value, determined for each launch configuration, increasing the spin rate will not reduce dispersion further. However, dispersion can be reduced further by rail launching techniques.

Wind effects on dispersion when all motors function properly are found to be virtually linear with variations in wind structure. This linearity is the dispersion in path angle at burnout. Typically, the total range for a $2\sigma$ wind is twice that of a $1\sigma$ wind. In assessing errors caused by winds, the degree to which winds can be measured must be considered.

For test conditions at Mach 3 to 5 and 140,000 to 150,000 ft, experience has shown launch-vehicle performance must be examined carefully. Such

tests require close attention to the vehicle design.  Dynamic character-
istics of the vehicle must be considered; for example, balancing the ve-
hicle and precisely controlling launching.

## 5.  HEAT TRANSFER

In establishing decelerator criteria for a Mars entry capsule, heat trans-
fer calculations delineate design and trajectory programming require-
ments.  Critical test-vehicle components and booster must be guarded
from excessive aerodynamic heating.  Velocities of the various stages
in significant atmospheric regions should be programmed to prevent ex-
cessive heating periods, thus minimizing the heat-protection require-
ments.  Frequently, minimum coasting periods between stages in multi-
ple-stage configurations can reduce heating, particularly since higher
Mach numbers will be encountered during ascent.  Adequate heat-protec-
tion materials must be provided for various exposed components of the
launch vehicle.  The maximum skin temperature allowable for most solid
propellant motors is determined by the temperature limits of propellant-
to-case bonding.  Fortunately, booster side area heating is reduced by
the relatively large distances from the nose.  Analyses and tests have
shown that wide ranges of skin temperatures are possible by varying
coasting times between stages.  These ranges can vary by as much as a
factor of four for large variations in coast times.  Mach number usually
is not significantly reduced by increasing the coast period; hence, coast
optimization to limit heating will not affect seriously overall perform-
ance.  However, substantial altitude variation can be expected with varia-
tions of the coast time.

Heating on inclined surfaces and at stagnation points also must be con-
sidered.  Heating at these points is more severe than side heating, and
appropriate selections must be made of leading-edge and nose radii,
material types, and material thicknesses.  Heat sink weight many times
is not practical and lightweight insulation must be provided for critical
areas.

## 6.  TEST VEHICLE CONCEPT

### a.  General

The test-vehicle concept consists of an aerodynamically "clean" pay-load-carrier forebody carrying the expandable decelerator test item in a container mounted in the after part of the carrier.  A deployable inflatable AIRMAT[a] cone envelopes the test item and, just prior to the test, gives the forebody its increased size, simulating the con-figuration and performance characteristics of the Mars Voyager en-try capsule.  A few seconds before reaching the test point, the AIR-MAT cone is inflated.

Figure 104 shows the basic suggested operational sequence of events on the mission trajectory.  The launch vehicle combines an XM-33 booster and two Recruit boosters for the first stage and a Lance booster for the second stage.

### b.  AIRMAT in Simulation

An inflatable AIRMAT cone simulates the basic entry capsule.  A multisided regular pyramid is selected for money-saving simplicity of fabrication and because it forms a reasonable approximation of a 120-deg blunt cone currently considered for the Mars entry capsule. Analysis shows that for a 12-ft diameter decagonal base, the maxi-mum deviation from a true circular base is three inches.  For low inflation pressures required at desired Mach number and altitude test conditions, operational loadings cause structural deflections of much less than an inch.

The design is based on a cone to be deployed at approximately 150,000 ft, in a dynamic pressure field ranging to about 60 psf.

The AIRMAT cone deployment Mach number, dynamic pressure, and altitude are anticipated to have the following ranges for two conditions:

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[a]TM, Goodyear Aerospace Corporation, Akron, Ohio.

    1.   $M \approx 4.0$; $q \approx 25$ psf; $h \approx 150,000$ ft

    2.   $M \approx 6.0$; $q \approx 60$ psf; $h \approx 155,000$ ft

Typical selection of materials for the AIRMAT cone is based on a flat-plate wall temperature of 500 to 600 F for an emissivity of 0.8. Consequently, Nomex or Dacron or both with heat protective coating are required.

The suggested skirt configuration is a regular 18-sided pyramid with ribs and flat sides. Both ribs and sides will have AIRMAT construction.

With asymmetric external aerodynamic loading, webs acting as spokes prevent distortion of the near-circular cross section of the skirt. The webs also prevent lateral displacement of the skirt with respect to the major longitudinal axis of symmetry.

The inner and outer skins forming the exterior skirt surfaces terminate in a welt clamped in place by a retainer ring. The ring eliminates the need for fastener holes through the fabric and permits uniform transfer of tensile loads. Pressure scaling of these interfaces can be accomplished easily during assembly.

c.   Design Considerations

Preliminary calculations indicate that, for an inflation pressure of 20 psi maintained to retain the shape of the AIRMAT cone with nominal deflection, the total weight is about 45 lb. To account for heat insulation coating and structural design, the total weight is estimated to be 100 lb with a deflection of about 2 in. The deflection can be decreased by increasing inflation pressure, an action that will increase weight and decrease the factor of safety.

This general design arrangement can be optimized easily for a specific set of design requirements. The important design variables are (1) number of ribs, (2) face sheet material, (3) number of drop threads, and (4) AIRMAT thickness.

ADDPEP DRAG
ADDPEP RETRO

C-BAND TRANSPONDER
225-MC TRANSMITTER

SPIN RATE
≈ 66 RPM

3000 LB MAX
VEHICLE

TRACKING FLARES

SECOND OR THIRD
STAGE DEPENDING
ON LAUNCH
CONFIGURATION

PAYLOAD

XM77
LANCE

XM33
CASTOR

LAUNCH

BEACON TRANSPONDER ON
FIRST STAGE IGNITION
TRACKING FLARES
TELEMETRY ON
SEQUENCER ON

161

DE-SPIN JETS

BRAKES OR
ROCKETS

150,000 FT
MACH 5.0
CONE DEPLOYED

NOSE CONE OFF

BOOSTER SEPERATED
BY EXPLOSIVE RING

IN-FLIGHT (ELECTROMECHANICAL TIMER) - AD

LAST STAGE BOOSTER IGNITION
LAST STAGE BOOSTER SEPARATION, DRAG
TELEMETRY CALIBRATE
DE-SPIN JETS (RATE GYRO)
ON-BOARD CAMERAS (2)
STATIC AND DYNAMIC PRESSURE PORTS OI
CANISTER EJECTION

DATA TRANSMISSION - ADDPEP TELEMETRY

STATIC PRESSURE
DYNAMIC PRESSURE
DECELERATOR SHOCK
DECELERATOR DRAG
ACCELERATION
TEMPERATURES
BALLUTE INTERNAL PRESSURE
FM/FM PAM EVENT MONITORING

DECELERATION CANISTER OFF
BALLUTE DEPLOYED

DPEP PROGRAMMING

BRAKES

PEN

RECOVERY
PARA BALLUTE
DEPLOYED

WATER OR LAND IMPACT

PARA BALLUTE FLOTATION
240-MC BEACON
FLASHING LIGHT
DYE MARKER
37-KC MARKER
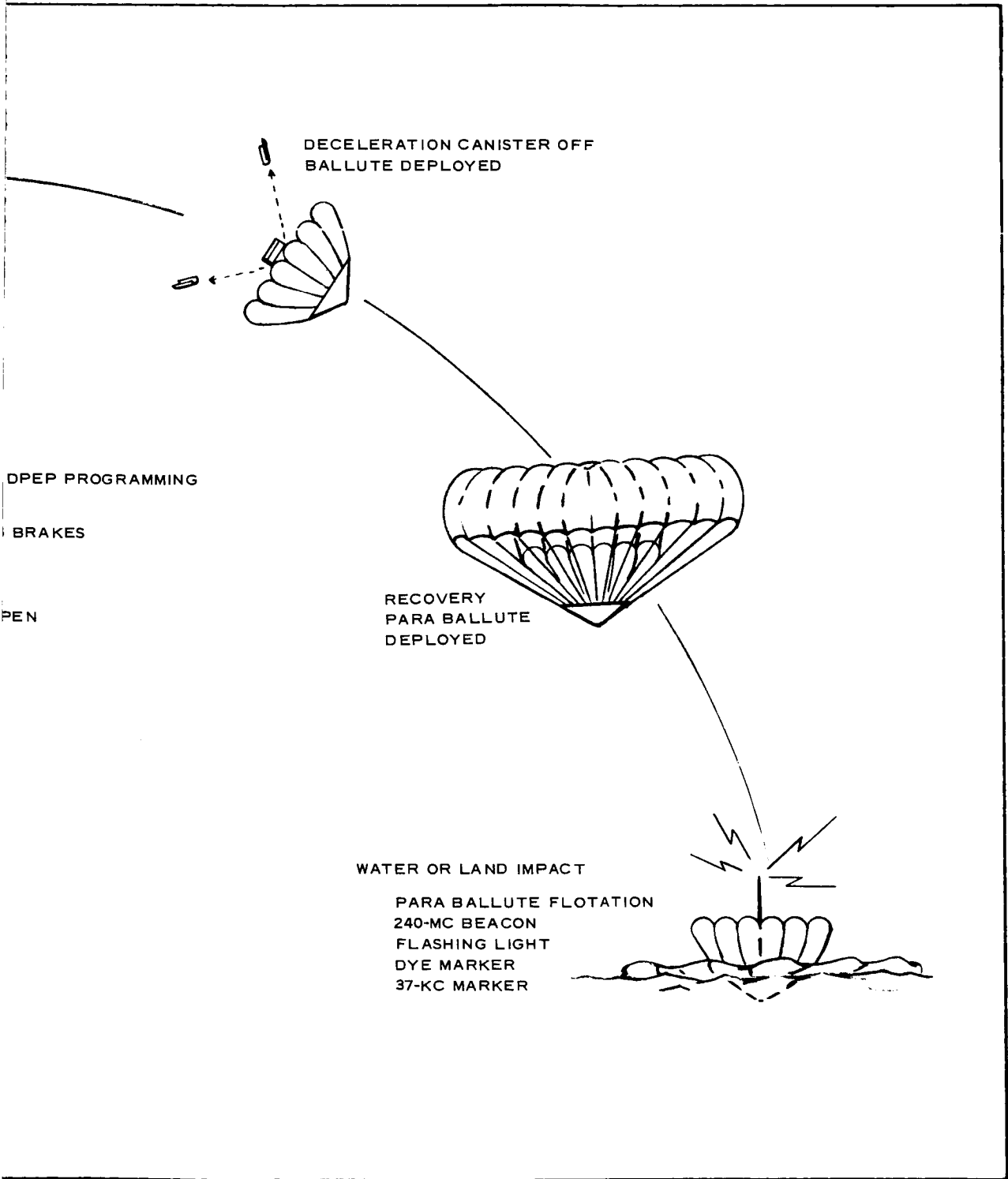
Figure 104 - Rocket Boost Techniques for Simulation Tests

If a conical shape is mandatory, the same basic arrangement can be used except that, instead of being flat the faces would be curved by tucking the inner-face in "pinch-pleats" or by slitting the outer face and masking the slits.

The most sophisticated and expensive alternate approach would entail actually weaving symmetrical, curved, tapered AIRMAT sections of the flare.

Again referring to the Air Launched, Air Recoverable Rocket (ALARR) flared skirt discussed above, design and development produced these conclusions and recommendations:

1. Functional testing must be conducted to solidly support and validate the analyses made in determining the design.

2. The conical shape, cantilevered extension beyond hard superstructure, and the sharp-cornered after termination at the base combine to provide a good aerodynamic configuration.

3. Inflation tests and asymmetric loading of the skirt indicate that the flare could be considered a rigid body under dynamic pressure and angle of attack environment.

4. Altitude-chamber inflation tests have demonstrated gas storage bottles, manifold, and valving are adequate for completely inflating a decelerator in a fraction of a second.

5. Structural integrity of fabric intersections of low and high-temperature skirts has been demonstrated under test conditions duplicating mission environments.

6.   Skirt packaging in a annular container similar to that suggested in this section has been demonstrated satisfactorily by Goodyear Aerospace at Holloman Air Force Base, N. M.

7.   A demonstration test at the Holloman AFB solid-track facility resulted in successful deployment, inflation, and erection of the skirt system.

7.   INSTRUMENTATION SYSTEM

The major areas of consideration for instrumentation of the Mars entry capsule decelerator test vehicle include (1) requirements, (2) environments, (3) unique conditions, and (4) survey.

In government programs for expandable aerodynamic decelerator development, free-flight test data have led to development and selection of instrument systems appropriate to the Mars entry problem. Selection and optimization of components and systems usually was contingent upon results in each of the four areas. Various inputs were integrated to establish specific selections. These inputs became guidelines to support required compromise.

Major areas of investigation are outlined below to depict more directly the specific efforts of concentration that should be studied.

1.   Requirements

a.   Programming - data acquisition and recovery sensing and sequence

b.   Data acquisition

(1)   Electronic

Sensors - opening shock force, drag force, acceleration, test vehicle motion (inertial measuring technique), static pressure, density (ionization techniques), impact pressure, and temperature

Data recording and transmission - telemetry, magnetic tape, stored and delayed telemetry, and direct

Photographic - a separate discussion

c.  Timing and time synchronization

d.  Radar tracking

e.  Destruct systems

f.  Search and location

g.  Power

2.  Environments

a.  Altitude

b.  Acceleration

c.  Temperature

d.  Aerodynamic heating

3.  Unique conditions

a.  Measurement by decelerator attachment influence on shock and drag force

(1)  Strain gage sensors - wire and semiconductor

(2)  Capacity, inductance, and displacement sensors

b.  Temperature and thermal effects

(1)  Decelerator integrity - surface temperature, temperature gradients, and thermal energy absorption

(2)  Structural integrity - aerodynamic heating

c.  Altitude, density, pressure, and temperature

d.  Search beacons - pulsed and CW systems

4.  Surveys

a.  Evaluation of past and current systems, subsystems, and components

(1)  ADDPEP

(2)  NASA free-flight tests

(3)  Other government-sponsored FFT programs

      b.   Item a devices indicating performance characteristics

           (1)  Environmental conditions encountered

           (2)  Accumulated statistical and tested reliability

           (3)  Operational history

      c.   Determination of prior concepts extended to present test objectives

           (1)  Direct integration

           (2)  Modification

           (3)  Radical change

      d.   Probability prediction for all suggested approaches and configurations

      e.   Reliability analysis

           (1)  Component selection

           (2)  System integration

           (3)  Redundancy

      f.   Probability prediction for given reliability versus economy

SECTION X - PRELIMINARY DESIGN, DEVELOPMENT, AND TESTING

1. OBJECTIVES

This section delineates work that must be accomplished for aerodynamic deceleration, stabilization, and terminal descent for a Mars entry capsule that ultimately will reach the planet's surface.

Such a deceleration system must slow the entry capsule and lower its effective ballistic coefficient to permit deployment of the terminal descent stage at a sufficiently high altitude.

Program objectives should include the following guidelines:

1. To establish engineering design details for the range of entry conditions specified and to determine the most effective means for reconfiguring and staging to achieve terminal velocity compatible with the capsule's surface-impacting subsystem

2. To establish engineering design details compatible with flight conditions

3. To establish analyses and tests

2. PRELIMINARY DESIGN

Point-mass trajectories should be computed for the vehicle deceleration system from primary decelerator deployment to surface impact. Initially, these computations should be selected from the specified entry trajectory furnished by the customer. Such combinations of initial point and decelerator ballistic coefficient must be chosen to achieve the velocity at altitude target specified by the customer. Histories of all significant flight and configuration variables for computed trajectories must be

included.  Results should be analyzed for vehicle and decelerator configurations, sets of initial conditions, atmospheric model, and velocity-at-altitude targets as are requested initially in the customer's Request for Quotation (RFQ).

3.  PRELIMINARY DECELERATOR ANALYSIS

Selection of the most promising decelerator should be based on preliminary aerodynamic, dynamic, structural, and design investigations. There investigations must be supported by examination of previous analyses.  Shapes and dimensions then would be determined to produce the required ballistic coefficients determined in Item 2 of this section.  The calculations and references used in selection must be reported and documented.

Transient thermal analysis must be conducted on primary decelerator fabrics before choices are made of fabrics and coatings that can withstand maximum calculated temperatures and heating conditions.

Pressure distribution over the primary decelerator must be determined for the operating range of Mach numbers.  Results would be used to establish an optimum fabricated shape after strength analysis described in this section.

Deployment force history, including shock loading, must be computed with gas-flow rates, reference areas, and drag characteristics.  The deployment sequence must be established, including time increments and limits.

The dynamic motion of the composite vehicle and decelerator be determined.  Acceleration and angular motions must be determined through evaluation similar in scope to that described for the point-mass trajectories.

Detailed preliminary engineering design drawings showing the decelerator, packaged and deployed, must be provided to the customer.  These

would include detailed construction features.  The equipment required for deployment sequence also must be presented.

Dynamic loads on the decelerator must be determined.  The snatch load of the vehicle and decelerator connector would be based on prior testing of similar decelerators.  After all loads are determined, critical or significant components must be analyzed structurally.

Decelerator weight must be determined according to structural analysis and geometric arrangement.

4.  DEVELOPMENT PROGRAM

a.  Five Phases

A development program could be divided into five phases: (1) analysis of preliminary design, (2) scale-model design and fabrication, (3) model testing, (4) full-scale system design, and (5) fabrication and flight testing of a full-scale system.

Work should be organized to provide maximum design data and to permit interphase branching.

Scheduled progress reporting during each phase with final reporting at the conclusions of each phase would provide suitable documentation.

After Phase V, a complete program report must be provided.

b.  Phase I - Test Evaluation of Decelerator Models

Wind tunnel tests should be conducted on a solid model of the decelerator shape established in preliminary design.  Tests should provide pressure distribution, drag coefficient, and moment analysis of the models at simulated Martian dynamic pressure and at atmospheric density.  A solid, small-scale model should be fabricated and instrumented for measurement of moment and pressure distribution in wind-tunnel tests.  Tests should duplicate dynamic loading and

density extremes of the projected Martian atmosphere as closely as
possible.

c.    Phase II - Test Evaluation of Larger-Scale Model

A model of approximately four-foot overall diameter should be de-
signed and fabricated reflecting Phase I results and analytical studies
of the preliminary design. Wind-tunnel tests, at simulated Martian
dynamic and atmospheric pressures, should be conducted to evaluate
deployment, stability, drag-time history, and general design.

d.    Phase III - System Design and Mockup Testing

A decelerator system incorporating recommended modification of the
Phase II configuration should be engineered and designed.

System engineering analysis must ensure decelerator system com-
patibility with the entry capsule and spacecraft bus. All flight en-
vironments including sterilization cycling will be included in these
analyses. After analysis, a complete decelerator system design
should be made and complete working drawings prepared.

A full-scale functional mockup of this design then should be fabri-
cated, with sufficient detail and durability to be used for design aid,
for evaluating packaging and deployment, and for drop testing in
Phase IV.

The mockup should be tested in a laboratory to determine the best
packaging and deployment techniques. Mockup dimensions should be
checked to verify fabrication procedures.

A report should delineate the work performed and solutions to prob-
lems encountered in Phase III.

e.    Phase IV - Air-Drop Test

The functional mockup, incorporating the best packaging and deploy-
ment design determined in Phase III, should be dropped from a C-130

aircraft.  Decelerator deployment staging should be accomplished at preset dynamic pressure.

Instrumentation, checkout, and ground-handling equipment should be fabricated and the functional mockup tested for operational readiness before shipment to the test site.  Controls should be provided for deploying each decelerator stage at the desired velocity.

The mockup should be dropped from an aircraft to examine deployment, inflation, and correlation with model test data.  Data on deployment, deceleration, and stability should be recorded and evaluated and a report prepared encompassing work conducted in Phase IV.

f.    Phase V - Flight Test

A flight test plan should establish full-scale balloon-launch testing and reduced-scale rocket-launch testing of the decelerator system at earth altitudes where the atmospheric density closely approximates that of the anticipated Mars entry environment.

Spacecraft and AGE must be fabricated to support the flight test plan.

A field-test crew must be assigned for assembly, checkout, and launch of the spacecraft.

Flight tests should be conducted according to the initial plan.  Flight test data must be recorded and evaluated.

A final report should be made of all program phases and final design for follow-on qualification and flight acceptance tests.

5.    SCHEDULE

A schedule for preliminary design and development (Figure 105) shows total work area in each phase with a number of stations for complete review of decelerator performance.  The schedule also provides firm checkpoints for control of schedule and expenditures.

The schedule permits complete program performance within 24 months,
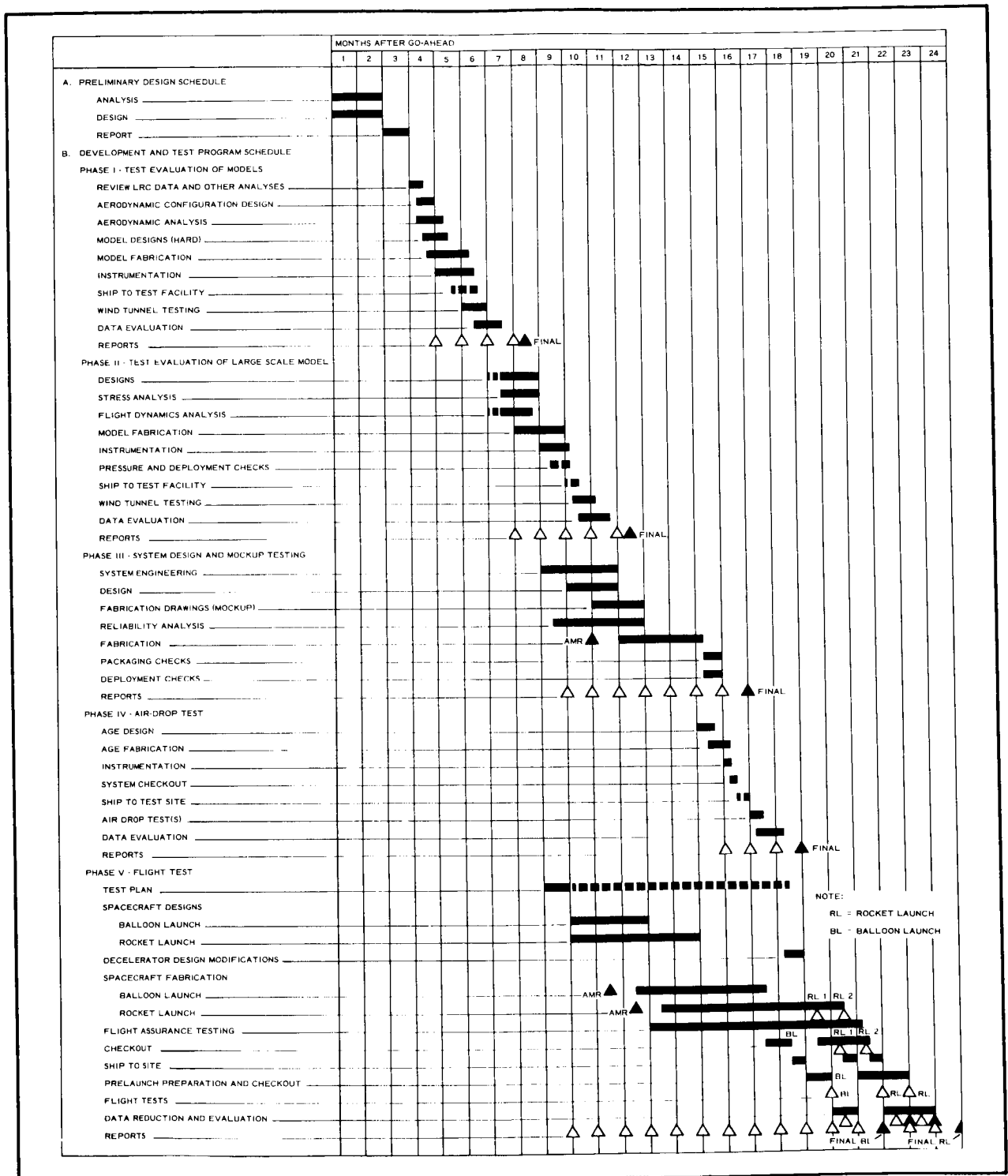
Figure 105 - Preliminary Program Schedule

making it compatible with the Voyager program freeze date for selection of configuration and subsystems.

PERT/time and PERT/cost controls should begin with Phase III. The PERT controls then would include the remainder of Phase II.

Technical briefings are scheduled at major technical decision points. More briefings could be added.

Field operations for the program include three distinct testing operations: wind-tunnel, air-drop, and flight.

Wind-tunnel testing could be performed at either NASA or Air Force facilities. An engineer associated with decelerator analysis should be assigned to supervise these tests, assisted by test technicians in setup and data recording.

Air-drop testing could be conducted with C-130 aircraft flown from Air Force Missile Test and Development Center, Holloman AFB, N.M. Radar and optical tracking and telemetry would record data. A decelerator systems test engineer as supervisor, an instrumentation and electrical engineer, and two test technicians should make up the field crew.

Flight testing could be conducted at the White Sands Missile Range. Because of the small number of flights, the same crew should conduct rocket and balloon launches. A field test manager, a rocket-launch program engineer, a balloon launch program engineer, an instrumentation and electrical engineer, three technicians, and a reliability and quality-assurance engineer should make up the crew.

6. DOCUMENTATION

Informal technical status reports should be submitted monthly, reporting activities from the first of the preceding month through the first of the current month. These reports should include but not be limited to (1) a brief summary of work status at the start of the current period and (2) a

description of work performed during the preceeding month, including positive or negative results.

Formal technical reports should be submitted within 30 days after completion of each phase. These reports should contain such items as calculations, graphs, sketches, data, and photographs.

A final report, documenting and relating the results of the entire proposed program, should be submitted within 30 days after the end of the flight test phase. Operation and maintenance manuals should be written for all AGE equipment used in the flight test program. Test plans should be submitted for approval 45 days prior to the air-drop and flight-test programs.

Financial status reports should be submitted monthly by the contract administrator for the contractor.

7. CUSTOMER RESPONSIBILITIES

 a. <u>Preliminary Design Phase</u>

  It would be required that the customer furnish to the contractor the following constraints and parameters:

   1. Physical profile characteristic of the Mars atmosphere

   2. Entry vehicle shape, dimensions, and subassembly arrangements

   3. Maximum allowable deceleration during deployment of auxiliary aerodynamic device

   4. Ballistic Mars-atmosphere entry trajectory, to impact, of the specified entry vehicle in specified atmosphere or atmospheres, at the specified initial entry path angle

   5. Initial trajectory conditions at deployment (including dynamic amplitudes and frequencies)

6.  Target velocity-at-altitude combinations for
    significant sequencing events:  secondary de-
    celerator deployment, impact, for example.
    Items 5 and 6 might be determined jointly by
    the contractor and the customer

b.  Development Program

During the development program, use of the following or similar
government-owned facilities would be required:

1.  Phase I - four-foot Arnold-Von Karman facility
    wind tunnel A

2.  Phase II - sixteen-foot transonic PWT wind tun-
    nel

3.  Phase III - building space at Holloman MTDS
    for assembly and checkout of mockup,  C-130
    aircraft, and range services of radar and opti-
    cal tracking, telemetry recording, and data
    reduction

4.  Phase IV - building space at Holloman MTDS
    and Walker AFB for assembly and checkout of
    spacecraft; launch balloon, adapter, and balloon
    launch services at Walker AFB; launch rockets
    and adapter sections; rocket launcher and re-
    lated equipment; and range services of radar
    tracking, telemetry recording, and data re-
    duction

## LIST OF SYMBOLS

$A$ = area

$a$ = acceleration (g)

$A'_{s(i)}$ = area of any of three parts of BALLUTE

$C$ = coefficient

$c$ = specific heat of material (Btu/lb-deg F)

$\bar{c}$ = constant depending on gas composition

$C_{dB}$ = BALLUTE drag coefficient

$C_{dC}$ = cone drag coefficient

$C_{do}$ = reference drag coefficient

$C_D$ = drag coefficient

CGS = center of gravity of surface

$C_{IS}$ =

$C_{L_\alpha}$ = lift coefficient slope

$C_{M_\alpha}$ = moment coefficient slope

$C_m$ = pitching moment coefficient

$C_{m_q}$ = damping derivative due to pitch rate

$C_{m_{\dot\alpha}}$ = damping derivative due to angle of attack rate

$C_{N_\alpha}$ = normal force coefficient slope

$C_n$ = yawing moment coefficient

$C_p$ = pressure coefficient

$C_{Pq}$ = pressure coefficient times dynamic pressure

$d$ = diameter

$D_B$ = BALLUTE drag

$d_B$ = BALLUTE diameter

$d_C$ = cone diameter

$D_o$ = reference diameter

$D_T$ = total drag

$F$ = pole load

$FS$ = factor of safety

$f_f$ = fabric stress

$h$ = altitude

$I$ = inertia

$I_{CGS}$ = moment of inertia about system cg

$I_{vYYi}$ = volume mass moment of inertia about Y-Y

$I_{YYi}$ = surface mass moment of inertia about the Y-Y axis

$$K = \frac{nT_m}{p\pi R_B^2}$$

$K_c$ = meridian strength to weight ratio (working)

$K_f$ = fabric strength to weight ratio (working)

$K_m$ = material strength to weight ratio (ultimate)

$L$ = length

$m$ = riser length forebody diameter ratio

$M_D$ = deployment mach number

$N$ = total drag to JPL drag ratio

$N$ = exponent depending on gas composition

$n$ = number of meridians

$P, p$ = pressure (gage)

$P_L$ = local static pressure on surface

$P_I$ = internal pressure

$P_\ell$ = surface pressure

$P_o$ = stagnation pressure behind normal shock (atmospheres)

$Pv$ = pressure energy (ft-lb)

$P_\infty$ = free-stream static pressure

$q$ = dynamic pressure; heat flux ($Btu/ft^2$)

$\dot{q}$ = heat flux rate ($Btu/ft^2$-sec)

$\dot{q}_o$ = stagnation point heat flux rate, ($Btu/ft^2$-sec)

$q_W$ = ratio of wall to total enthalpy

$R$ = radius

$R_B$ = BALLUTE radius

$R_C$ = cone radius

$R_N$ = nose radius

$R_o$ = reference radius

$R_v$ = forebody base radius

$r$ = energy constant

$T$ = gas temperature (F)

$T_f$ = final maximum temperature (F)

$T_i$ = initial temperature

$T_\ell$ = surface temperature

$T_m$ = load in one meridian

$T_\infty$ = ambient temperature

$U_\infty$ = free-stream velocity divided by 10,000 pfs

$V, v$ = volume

$V_B$ = volume of BALLUTE

$W$ = weight

$W_B$ = weight of BALLUTE

$W_C$ = cone weight

$W_f$ = fabric weight

$W_R$ = weight of riser

$W_T$ = total system weight

w = propellant weight (lb)

$\overline{X}_{①}$ = centroidal distance of any of three BALLUTE parts

$x_o$ = distance

$\overline{X}_s$ = BALLUTE center of gravity

$\overline{X}_v$ = center of gravity of complete BALLUTE volume

$\overline{Y}_{①}$ = centroidal distance of any of three BALLUTE parts

$y_w$ = viscous wake half-width

$\gamma$ = average weight per equal foot of BALLUTE surface

$\rho_h$ = density at altitude

$\rho$ = surface density

$\theta_s$ = body surface inclination

$\alpha$ = angle of attack

$$\rho = \frac{F}{p\pi R_B^2}$$

$\gamma$ = unit weight

$\Delta$ = increment

$\gamma$ = specific heat ratio

$\Gamma$ = adiabatic lapse rate ($K^+$/km and R/1000 ft)

$\theta$ = local surface inclination angle

$\delta$ = thickness of material (ft)

$\tau$ = time (sec)

$\rho$ = density of material (pcf)

$\sigma$ = standard deviation

# LIST OF REFERENCES

1. Marvin, J. G.; and Deiwert, G. S.: Convective Heat Transfer in Planetary Gases. NASA TR R-224, July 1965.

2. Lees, L.: "Laminar Heat Transfer Over Blunt-Nosed Bodies at Hypersonic Flight Speeds, " Jet Propulsion, April 1956; 259-268.

3. Champney, W. B.; Athans, J. B; and Mayerson, C. D.: A Study of Hypersonic Aerodynamic Drag Devices. WADC TR 59-324, Part II. Wright-Patterson Air Force Base, Ohio. December 1960.

4. Alexander, W. C.: Investigation to Determine the Feasibility of Using Inflatable Balloon Type Drag Devices for Recovery Applications in the Transonic, Supersonic and Hypersonic Flight Regime. ASD-TDR-62-702, Part II. Wright-Patterson Air Force Base, Ohio. October 1962.

5. JPL Voyager Capsule Flight Equipment TA and FA Test Procedures For Heat Sterilization and ETO Decontamination Compatibility, Vol - 50503-ETS. Pasadena, Calif., Jet Propulsion Laboratory, California Institute of Technology.

6. Anderson, A. R.: Effect of Biological Sterilization and Vacuum on Certain Parachute Retardation System Components. Final Rep 4324. Pasadena, Calif., Jet Propulsion Laboratory, California Institute of Technology. August 1963.

7. Chernowitz, G.; and DeWeese, J. H.: Performance of and Design Criteria for Deployable Aerodynamic Decelerators. Second major revision of USAF Parachute Handbook, ASD-TR-61-579, AD-429971, December 1963.

8. ASD-TDR-62-542: New and Improved Materials for Expandable Structures. Pasadena, Calif., Jet Propulsion Laboratories, California Institute of Technology. June 1962.

9. AEDC TR-65-218: Drag Characteristics of Three BALLUTE Decelerators in the Wake of the ALARR Payload at Mach Numbers 2 to 4. Arnold Air Force Station, Tenn. Arnold Engineering Development Center. 1965.

10. AEDC TDR-63-119: Performance of Flexible Aerodynamic Decelerators at Mach Numbers from 1.5 to 6. Arnold Air Station, Tenn. Arnold Engineering Development Center. 1963.

11. Alexander, W. C.: Investigation to Determine the Feasibility of Using Inflatable Balloon Type Drag Devices for Recovery Applications in the Transonic, Supersonic, and Hypersonic Flight Regime. Part II. Mach 4 to Mach 10 Feasibility Investigation. Akron, Ohio. Goodyear Aircraft Corp. ASD TDR-62-704. December 1965.

12. Charczenko, N.: Aerodynamic Characteristics of Towed Spheres, Conical Rings and Cones Used as Decelerators at Mach Numbers from 1.57 to 4.65. TN D-1789. NASA, April 1963.

13. Charczenko, N.; and McShera, J. T.: Aerodynamic Characteristics of Towed Cones Used as Decelerators at Mach Numbers from 1.57 to 4.65. TN D-994. NASA, December 1965.

14. Altgelt, R.: Gemini BALLUTE System Development. GER-11538 Akron, Ohio. Goodyear Aerospace Corporation. January 1965.

15. GER-11665: Aerodynamic Deployable Decelerator Performance Evaluation Program. Akron, Ohio. Goodyear Aerospace Corporation.

16. AEDC-TR-65-266: The Characteristics of a BALLUTE in the Wake of the Martin Fullscale SV-5D Vehicle at Mach Numbers from 0.6 to 2.9. Arnold Air Station, Tenn. Arnold Engineering Development Center. 1965.

17. AEDC-TR-64-131: Gemini BALLUTE Structural Test at Mach Numbers 0.55 and 1.92. Arnold Air Station, Tenn. Arnold Engineering Development Center. 1964.

18. McShera, J. T.: Aerodynamic Drag and Stability Characteristics of Towed Inflatable Decelerators at Supersonic Speeds. TN D-1601. NASA.

19. GER-11762: BALLUTE Evaluation Program for Project Sleigh Ride. Akron, Ohio. Goodyear Aerospace Corporation.

## APPENDIX A - DETAILED ANALYSIS OF DECELERATORS

1. STRUCTURAL AND WEIGHT ANALYSES

   a   General

   Weight calculations for several fabric decelerators for a Mars entry capsule are contained in this appendix. Decelerators described are: trailing BALLUTEs; enveloping BALLUTEs, plain and tucked back; attached skirts; and AIRMAT cone. This work was done to assist in evaluation of these configurations and to assist in selection of designs to be studied in more detail.

   The weights of the fabric decelerators are functions of the drag increase, the deployment conditions, the vehicle size and weight, and the strength of the selected materials. No allowances are included for the container weights, ejection systems, construction details, and inflation system if used. The effects of these omitted items are to be included in the analysis of the selected designs.

   b.   Trailing BALLUTEs

   The weight equation[a] for a plain-back BALLUTE is

   $$W_B = P \frac{\pi}{8} d_B^3 \left[ \frac{4(1 - K)}{K_f} + \frac{2\pi K}{K_c} \right] , \qquad \text{(A-1)}$$

   where

   $$P = C_p q , \qquad \text{(A-2)}$$

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[a] Alexander, W. C.: Investigation to Determine the Feasibility of Using Inflatable Balloon-Type Drag Devices for Recovery Applications in the Transonic, Supersonic, and Hypersonic Flight Regimes, Part II, Mach 4 to Mach 10 Feasibility Investigation. ASD-TDR-62-702, December 1962, p 140.

$$q = \frac{aW_T}{C_{do}\pi \frac{(d_o)^2}{4}} \qquad (A-3)$$

$$D_o = C_{do}q\pi \frac{(d_o)^2}{4} , \text{ and} \qquad (A-4)$$

$$aW_T = C_{do}q\pi \frac{(d_o)^2}{4} . \qquad (A-5)$$

In these equations,

$P$ = pressure (gage),

$C_p$ = pressure coefficient,

$q$ = dynamic pressure,

$a$ = acceleration (g's),

$W_T$ = total system weight,

$C_{do}$ = reference drag coefficient,

$D_o$ = reference drag, and

$d_o$ = reference diameter.

The equation for vehicle drag with decelerator deployed, as a function of the basic vehicle drag, is

$$D_T = ND_o \qquad (A-6)$$

or

$$D_T = D_o + D_B , \qquad (A-7)$$

when

$D_T$ = total drag,

$$N = \text{total drag to JPL capsule drag ratio,}$$

$$D_B = \text{BALLUTE drag load, and}$$

$$D_o = \text{reference drag;}$$

then

$$D_B = (N - 1)D_o .\qquad\qquad(A-8)$$

BALLUTE drag is expressed as

$$D_B = C_{dB}\pi q \frac{(d_B)^2}{4}\qquad\qquad(A-9)$$

when

$$(d_B) = \text{BALLUTE diameter, and}$$

$$C_{dB} = \text{BALLUTE drag coefficient.}$$

With $C_{dC}$ = coefficient of cone drag, let

$$C = \frac{C_{dB}}{C_{do}}\qquad\qquad(A-10)$$

or

$$C = \frac{C_{dC}}{C_{do}} ;\qquad\qquad(A-11)$$

then, substitution in Equation A-9 and Equations A-4, A-8, and A-10 produces

$$(N - 1)\left[C_{do}\pi q \frac{(d_o)^2}{4}\right] = C\left(C_{do}\pi q \frac{d_B^2}{4}\right) ,$$

and

$$d_B = \sqrt{\frac{N - 1}{C}}\, d_o .\qquad\qquad(A-12)$$

Substitution in Equation A-1 the values from Equations A-2, A-5, and A-12 produces the equations:

$$W_B = \frac{C_p a W_T}{C_{do} \pi \frac{(d_o)^2}{4}} \left(\sqrt{\frac{N-1}{C}} d_o\right)^3 \frac{\pi}{8} \left[\frac{4(1-K)}{K_f} + \frac{2\pi K}{K_c}\right] ,$$

and

$$\frac{W_B}{a W_T} = \frac{C_p d_o}{2 C_{do}} \sqrt{\left(\frac{N-1}{C}\right)^3} \left[\frac{4(1-K)}{K_f} + \frac{2\pi K}{K_c}\right] , \qquad (A-13)$$

when

$$K = \frac{n T_m}{p \pi (R_B)^2} ,$$

$n$ = number of meridians,

$T_m$ = load on one meridian,

$R_B$ = BALLUTE radius,

$K_c$ = meridian strength-weight ratio (working), and

$K_f$ = fabric strength-weight ratio (working).

With an additional factor of safety (FS) on the riser and assuming the riser to be made of the same material as the meridians with a riser length of "m" JPL diameters, a weight equation for the riser becomes:

$$W_R = \frac{D_B m d_o (FS)}{K_c} . \qquad (A-14)$$

From Equations A-4, A-5, and A-8, where m = riser length fore-body diameter ratio,

$$D_B = a W_T (N-1) , \qquad (A-15)$$

$$W_R = \frac{aW_T(FS)md_o}{K_c}(N-1) \ ,$$

and

$$\frac{W_R}{aW_T} = \frac{(FS)(m)(d_o)}{K_c}(N-1) \ . \tag{A-16}$$

The total weight of the BALLUTE and riser is found by adding Equations A-13 and A-16.

$$\frac{W_B + W_R}{aW_T} = d_o(N-1)\left\{ \frac{C_p}{2C_{do}}\sqrt{\frac{N-1}{C^3}}\left[ \frac{4(1-K)}{K_f} + \frac{2\pi K}{K_c} \right] + \frac{FSm}{K_c} \right\} \ . \tag{A-17}$$

Equation A-17 gives the weight of a trailing BALLUTE and riser line as a function of the deployment conditions, basic body diameter and weight, the desired drag increase, and BALLUTE material.

c.  **Enveloping BALLUTEs**

(1)  Plain-Back BALLUTE

The weight of a plain-back BALLUTE is given by Equation A-1. Equations A-2 through A-8 also are applicable to plain-back enveloping BALLUTEs. The additional drag from the BALLUTE is given by the following equation,

$$D_B = C_{dB}\pi q\, \frac{(d_B)^2 - (d_o)^2}{4} \ . \tag{A-18}$$

Substitution of Equations A-4, A-10, and A-16 into A-8 yields

$$C(C_{do}\pi q)\, \frac{(d_B)^2 - d_o^2}{4} = (N-1)C_{do}q\pi\, \frac{d_o^2}{4} \ ,$$

$$C(d_B^2 - d_o^2) = (N-1)d_o^2 \ ,$$

and

$$d_B = \sqrt{\frac{(N + C - 1)}{C}} \, d_o \, . \qquad (A-19)$$

Substitution from Equations A-2, A-5, and A-19 into Equation A-8 produces:

$$W_B = \frac{C_p a W_T}{C_d \pi \frac{d_o^2}{4}} d_o^3 \sqrt{\left(\frac{N + C - 1}{C}\right)^3} \frac{\pi}{8} \left[\frac{4(1 - K)}{K_f} + \frac{2\pi K}{K_c}\right]$$

and

$$\frac{W_B}{a W_T} = \frac{C_p d_o}{2 C_{do}} \sqrt{\left(\frac{N - 1}{C} + 1\right)^3} \left[\frac{4(1 - K)}{K_f} + \frac{2\pi K}{K_c}\right] \, . \qquad (A-20)$$

(2)  Tucked-Back BALLUTEs

The weight equation for a tucked-back BALLUTE, whose configuration fits the JPL capsule sketched in JPL Drawing J-141989, is developed in Item 2 of this appendix.

The equation is

$$W_B = P\pi \frac{(d_B)^3}{8} \left[\frac{1.29(0.537 - K)}{K_f} + \frac{2.15 K}{K_c}\right] \, . \qquad (A-21)$$

Substitution of Equations A-2, A-5, and A-19 into Equation A-21 gives:

$$W_B = \frac{C_p a W_T}{C_{do} \pi \frac{d_o^2}{4}} d_o^3 \sqrt{\left(\frac{N - 1}{C} + 1\right)^3} \frac{\pi}{8} \left[\frac{1.29(0.537 - K)}{K_f} + \frac{2.15 K}{K_c}\right],$$

and

$$\frac{W_B}{a W_T} = \frac{C_p d_o}{2 C_{do}} \sqrt{\left(\frac{N - 1}{C} + 1\right)^3} \left[\frac{1.29(0.537 - K)}{K_f} + \frac{2.15 K}{K_c}\right] \, . \quad (A-22)$$

BALLUTE weight is a function of deployment conditions, material properties, total capsule weight, and drag ratio; that is, total drag to JPL drag.

d. AIRMAT Cone Decelerator

A weight equation for an AIRMAT cone decelerator with a half-cone angle of $\theta$ and a half-taper angle of $\phi$ is developed in Item 2.

The equation is:

$$W_C = \frac{6(FS)P}{K_M} \frac{\pi}{8} \left(\frac{\cos \phi}{\sin \theta}\right)^2 \cos \phi (\tan \theta - \tan \phi)(d_C^3 - d_o^3) , \quad (A-23)$$

where

$K_M$ = material strength-weight ratio (ultimate), and

$d_C$ = cone diameter.

Substitution from Equations A-2, A-5, and A-19 into Equation A-23 gives the weight of the cone as a function of deployment conditions, material properties, total weight of the capsule, and drag ratio; that is, total drag to JPL drag. The equation becomes

$$\frac{W_C}{aW_T} = \frac{3(FS)C_p d_o}{K_M C_{do}} \left(\frac{\cos \phi}{\sin \theta}\right)^2 \cos \phi (\tan \theta - \tan \phi) \left[\left(\sqrt{\frac{N-1}{C} + 1}\right)^3 - 1\right] .$$

$$(A-24)$$

e. Decelerator Comparison

Equation A-17 is evaluated for two cases of trailing BALLUTEs. Numerical values are listed below:

For a riser four diameters in length, values are:

C = 0.8,

$C_{do}$ = 1.45,

$C_p$ = 2.75,

$d_o = 12$ ft,

$FS = 2.0$,

$K = 0.6$,

$K_c = 24,900$ ft $\left.\right\}$ 600 F Nomex      37,500 ft $\left.\right\}$ 350 F Dacron, and

$K_f = 17,700$ ft $\left.\right.$                      26,700 ft $\left.\right.$

$m = 4.0$.

For a riser eight diameters in length, values are

$C = 0.9$,

$C_{do} = 1.45$,

$C_p = 2.75$,

$d_o = 12$ ft,

$FS = 2.0$,

$K = 0.6$,

$K_c = 24,900$ ft $\left.\right\}$ 600 F Nomex      37,500 ft $\left.\right\}$ 350 F Dacron, and

$K_f = 17,700$ ft $\left.\right.$                      26,700 ft $\left.\right.$

$m = 8.0$.

Solving Equation A-17 for BALLUTE fabric weight as a function of total BALLUTE weight produces the equation,

$$\frac{W_f}{W_B + W_R} = \frac{\left[\sqrt{\dfrac{N-1}{C^3}}\ \dfrac{4(1-K)}{K_f}\right]\dfrac{C_o}{2C_{do}}}{\dfrac{C_p}{2C_{do}}\left[\sqrt{\dfrac{N-1}{C^3}}\right]\left[\dfrac{4(1-K)}{K_c} + \dfrac{2\pi K}{K_c}\right] + \dfrac{(FS)m}{L_c}} . \quad (A-25)$$

Equation A-12 in nondimensional form becomes

$$\frac{d_B}{d_o} = \sqrt{\left(\frac{N - 1}{C}\right)} \; . \tag{A-26}$$

Equations A-25 and A-26 also can be evaluated for two types of trailing BALLUTEs (see Figures A-1 and A-2).

Equation A-20 can be evaluated for a plain-back BALLUTE using the following data:

$$C = 0.90,$$

$$C_{do} = 1.45,$$

$$C_p = 2.75,$$

$$d_o = 12 \text{ ft},$$

$$K = 0.6, \text{ and}$$

$$K_c = 24,900 \text{ ft} \left.\vphantom{\begin{matrix}a\\b\end{matrix}}\right\} 600 \text{ F Nomex} \qquad \begin{matrix} 37,500 \text{ ft} \\[8pt] 26,700 \text{ ft} \end{matrix} \left.\vphantom{\begin{matrix}a\\b\end{matrix}}\right\} 350 \text{ F Dacron.}$$
$$K_f = 17,700 \text{ ft}$$

Equation A-19 becomes, in nondimensional form:

$$\frac{d_B}{d_o} = \sqrt{\frac{N - 1}{C} + 1} \; . \tag{A-27}$$

Solving Equation A-20 for BALLUTE fabric weight as a function of total BALLUTE weight produces the following equation (see Figure A-3),

$$\frac{W_f}{W_B} = \frac{\dfrac{4(1 - K)}{K_f}}{\dfrac{4(1 - K)}{K_f} + \dfrac{2\pi K}{K_c}}$$

$$= 0.38 \; .$$

Equation A-22 for a tucked-back BALLUTE is evaluated using the following data:
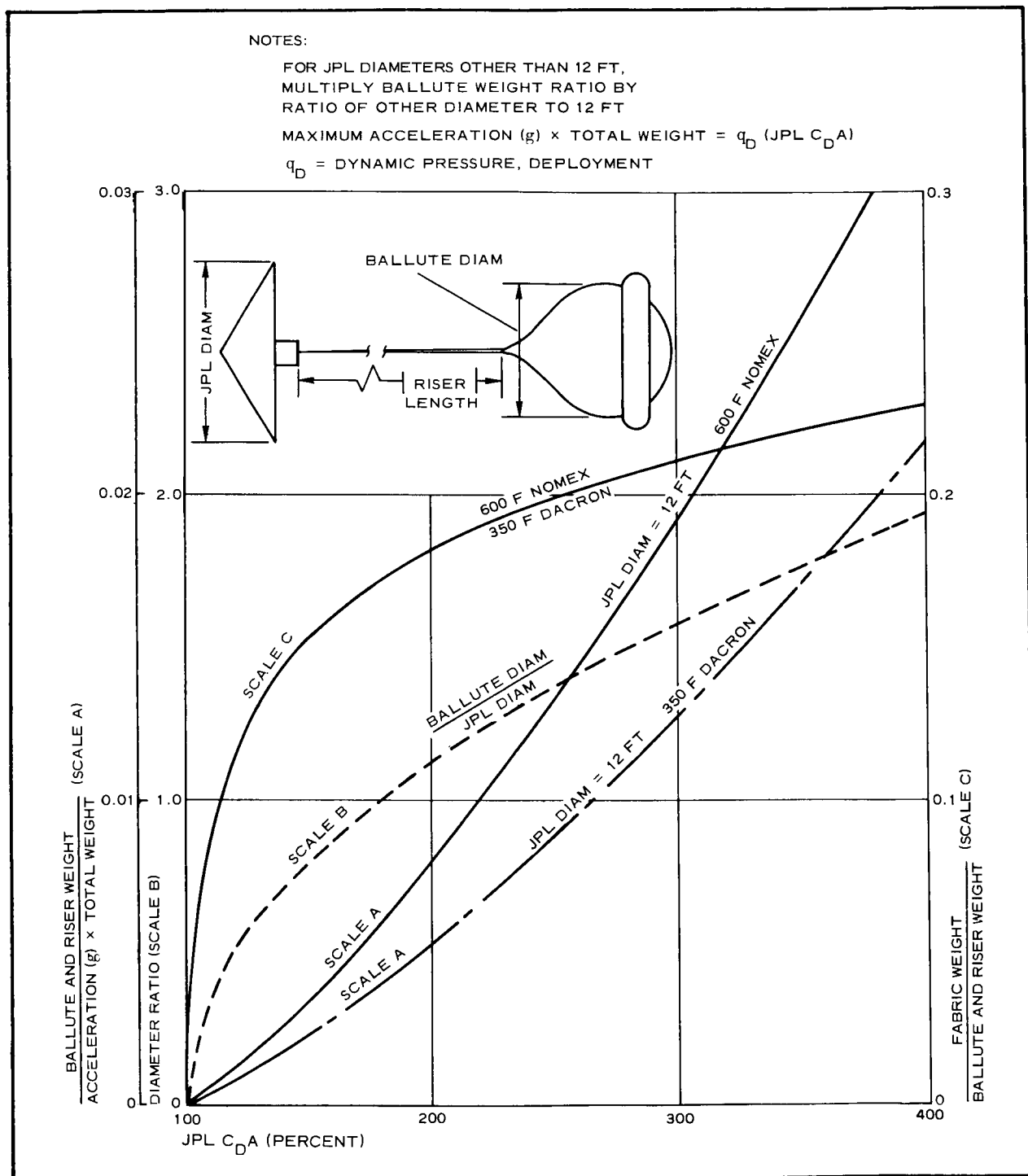
Figure A-1 - Trailing BALLUTE Weight, Ram-Air Inflation, Riser
Length of Four JPL Diameters
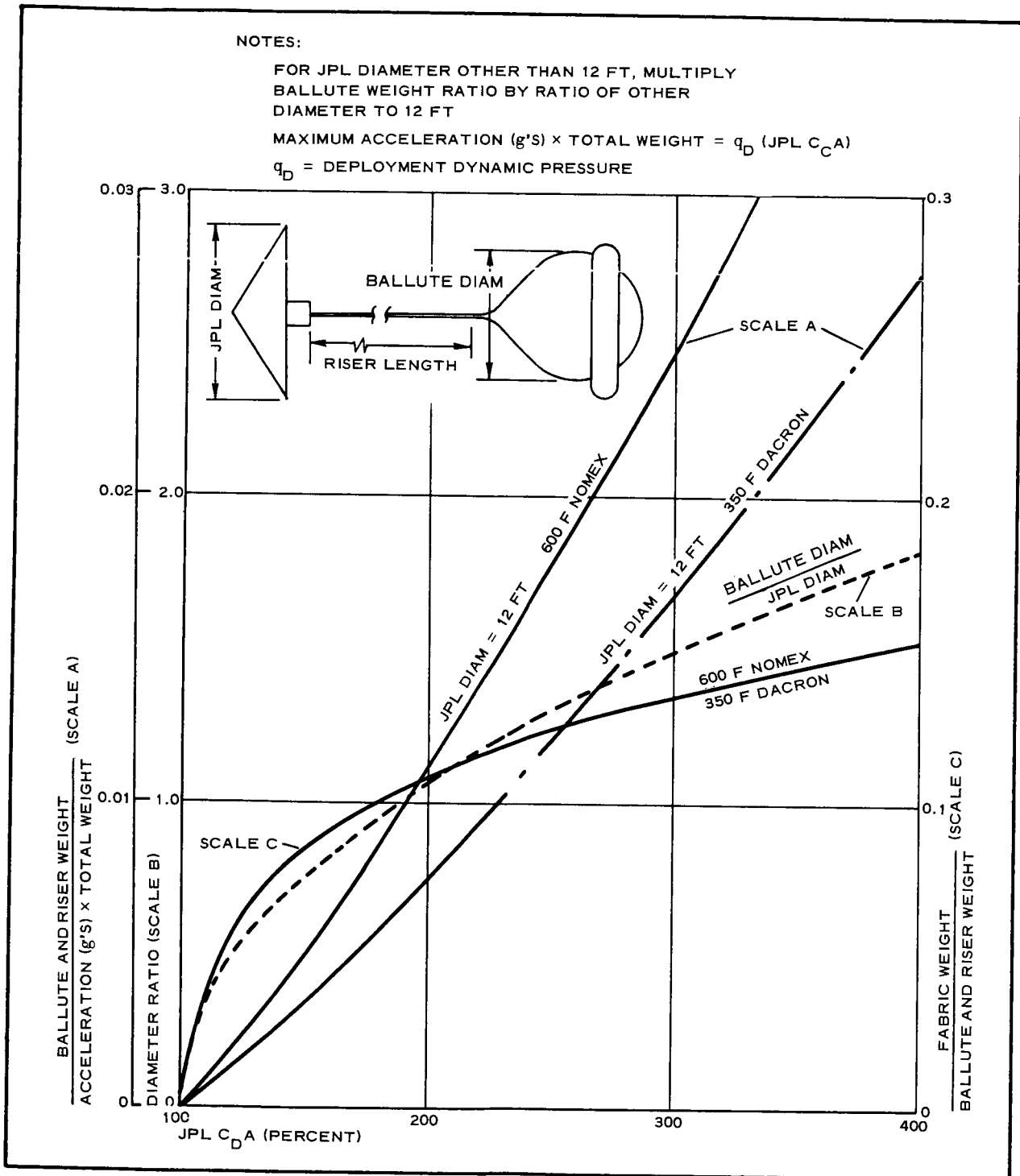
Figure A-2 - Trailing BALLUTE Weight Ram-Air Inflation, Riser
Length of Eight JPL Diameters

NOTES:

FOR JPL DIAM OTHER THAT 12 FT, MULTIPLY
BALLUTE WEIGHT RATIO BY RATIO OF OTHER
DIAM TO 12 FT

MAXIMUM ACCELERATION (g) × TOTAL WEIGHT = $q_D$ (JPL $C_D A$)

$q_D$ = DEPLOYMENT DYNAMIC PRESSURE

FABRIC WEIGHT = 38 PERCENT OF BALLUTE WEIGHT
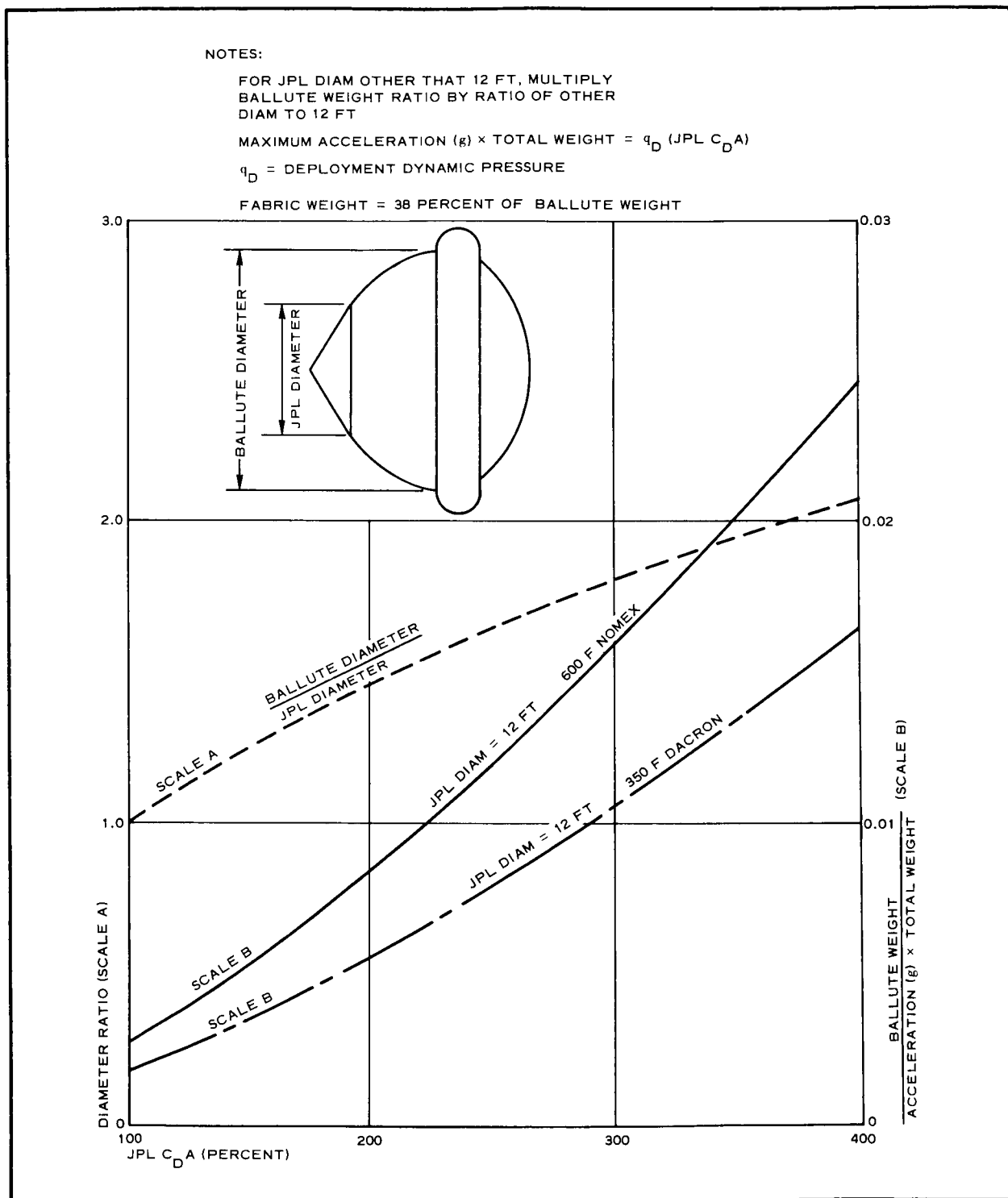
Figure A-3 - Enveloping BALLUTE Weight with Ram-Air Inflation

$$C = 0.61,$$

$$C_{do} = 1.45,$$

$$C_p = 2.75.$$

$$d_o = 12 \text{ ft,}$$

$$K = \rho = 0.475. \text{ and}$$

$$\left. \begin{aligned} K_c &= 24,900 \text{ ft} \\ K_f &= 17,700 \text{ ft} \end{aligned} \right\} 600 \text{ F Nomex} \quad \left. \begin{aligned} 37,500 \text{ ft} \\ 26,700 \text{ ft} \end{aligned} \right\} 350 \text{ F Dacron,}$$

where

$$\rho = \frac{F}{P\pi R_B^2} \ .$$

When Equation A-27 is used for the diameter ratio, solving Equation A-22 for BALLUTE fabric weight as a function of total BALLUTE weight produces the equation (see Figure A-4),

$$\frac{W_f}{W_B} = \frac{\dfrac{1.29(0.537 - K)}{K_f}}{\dfrac{1.29(0.537 - K)}{K_f} + \dfrac{2.15\,K}{K_c}}$$

$$\cong 0.10 \ .$$

Equation A-24 for the AIRMAT cone decelerator can be evaluated using the following data (see Figure A-5),

$$C = 1.0,$$

$$C_{do} = 1.45,$$

$$C_p = 1.45,$$

$$d_o = 12 \text{ ft,}$$

$$FS = 2.0,$$

NOTES:

FOR JPL DIAMETER OTHER THAN 12 FT MULTIPLY
BALLUTE WEIGHT RATIO BY RATIO OTHER
DIAMETER TO 12 FT

ACCELERATION (g'S) × TOTAL WEIGHT = $q_D$ (JPL DIAMETER)

$q_D$ = DEPLOYMENT DYNAMIC PRESSURE

FABRIC WEIGHT = 10 PERCENT BALLUTE WEIGHT
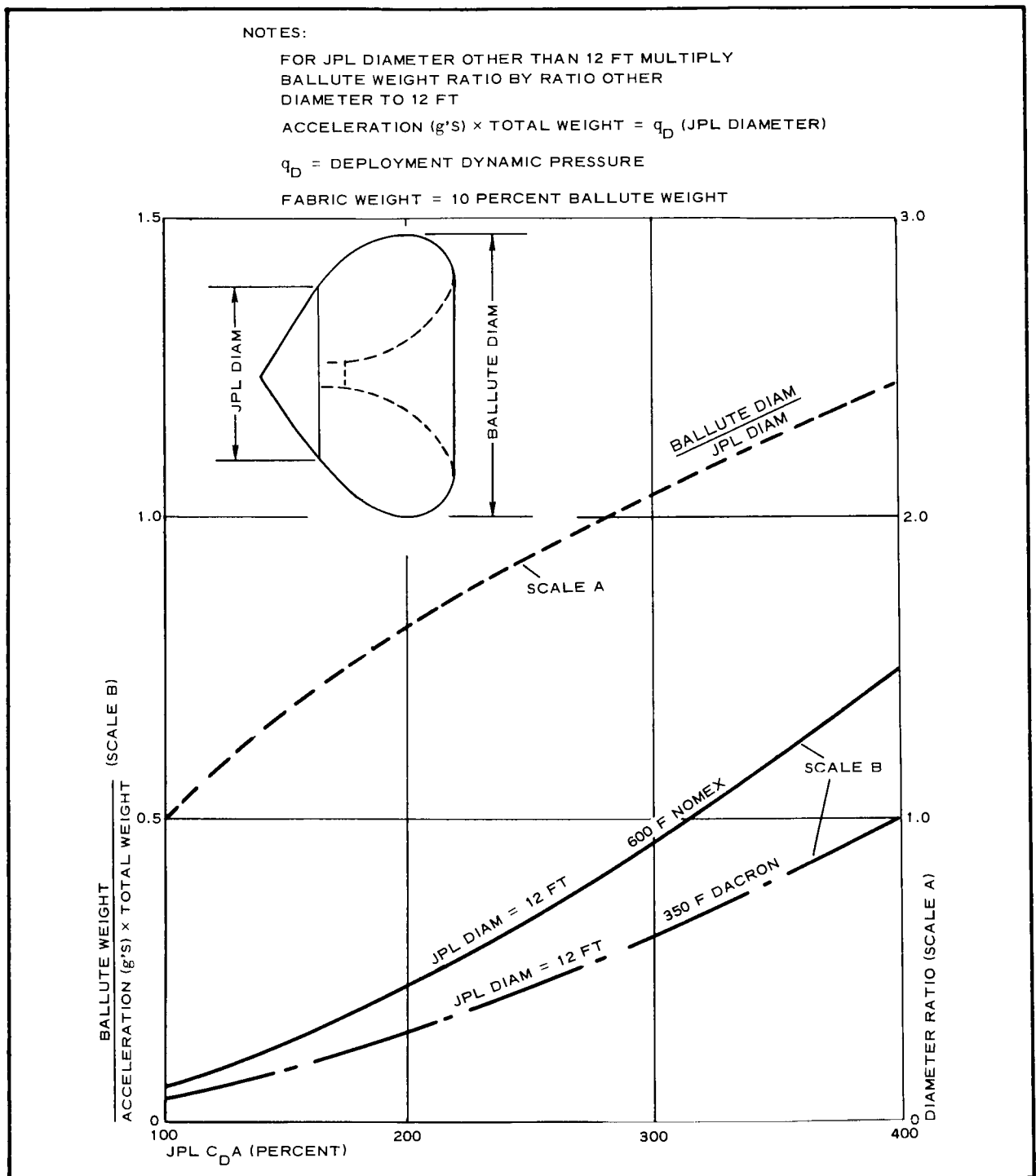


Figure A-4 - Enveloping BALLUTE Weight, Ram-Air Inflation,
Tucked-Back Without Burble Fence

NOTES:

FOR JPL DIAMETER OTHER THAN 12 FT, MULTIPLY
CONE WEIGHT RATIO BY RATIO OF OTHER DIAMETER
TO 12 FT DROP

ACCELERATION (g'S) × TOTAL WEIGHT = $q_P$ (JPL $C_D A$)

$q_P$ = DEPLOYMENT DYNAMIC PRESSURE

DROP THREAD WEIGHT $\cong$ 1/3 CONE WEIGHT
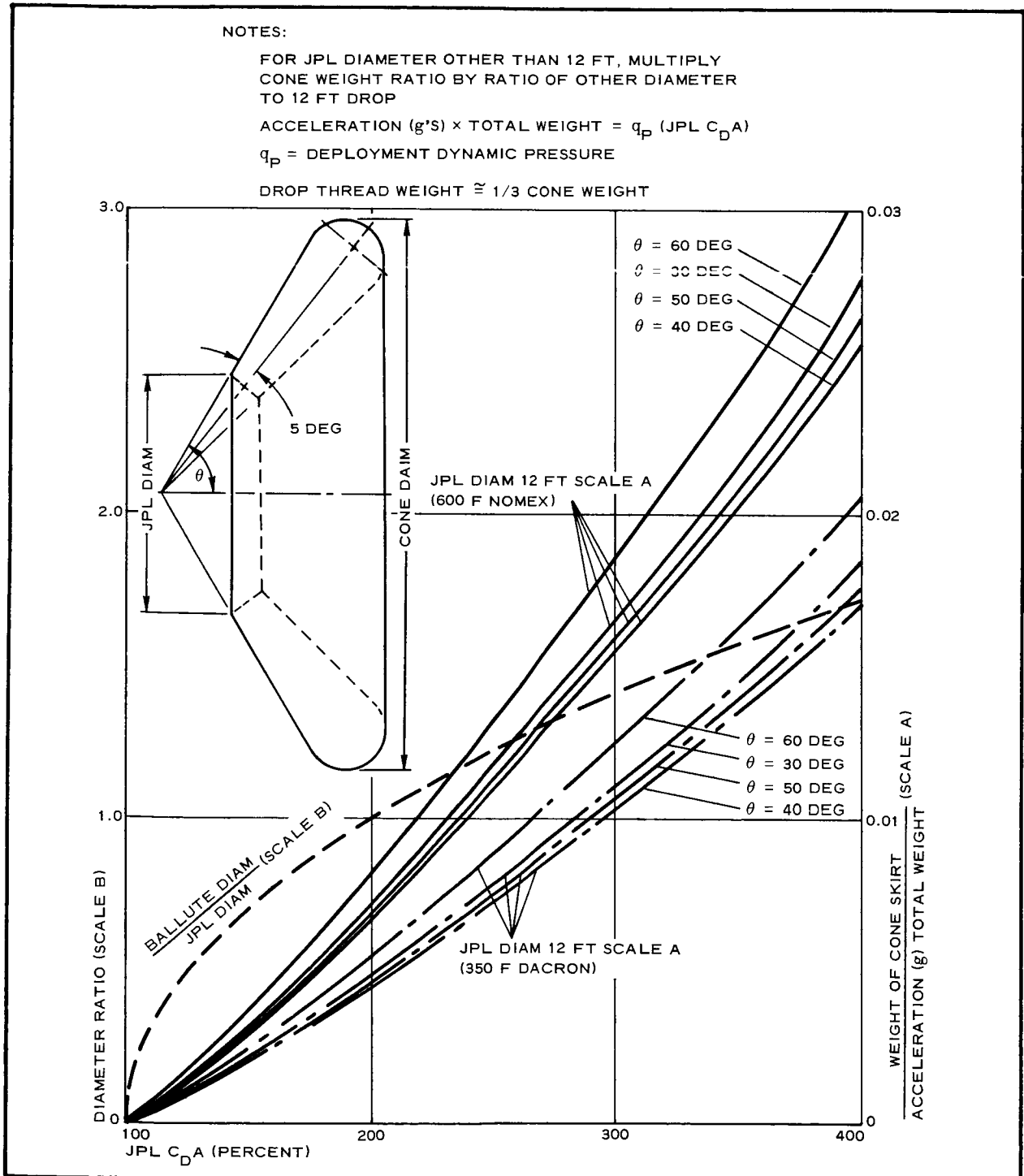


Figure A-5 - Conical AIRMAT Skirt Weight Analysis
(without Inflation-System Weight)

$$K_M = 35,000 \text{ ft (600 F Nomex)}, \ 52,800 \text{ ft (350 F Dacron)},$$

$$\phi = 5 \text{ deg, and}$$

$$\theta = 60, \ 50, \ 40, \text{ and } 30 \text{ deg.}$$

## 2.  WEIGHT EQUATIONS

### a.  Tucked-Back BALLUTE

The weight equations for the tucked-back BALLUTE, whose configuration fits the sketched shape of the JPL capsule, have been calculated by selecting isotensoid curves that fit a small-scale layout of the capsule back.[a] The resulting shape has a drag ratio of about 2.25. Although the weight equations are for this specific design, the drag ratio is near the middle of the range of interest. Therefore, these weight equations are considered general weight equations for this decelerator. The profile used in developing the equations is presented to scale in Figure A-6. In this illustration;

$$\rho = \frac{F}{P\pi(R_B)^2},$$

$$F = \text{pole load, and}$$

$$L_M = \text{meridian length.}$$

Thus, the centroid of meridian length becomes

$$X_L = \frac{(0.143 R_B)(9.0142 R_B)}{2.145 R}$$

$$= 0.601 R_B. \tag{A-28}$$

---

[a]Houtz, N. E.: "Optimization of Inflatable Drag Devices by Isotensoid Design," AIAA Paper 64-437.

| INCREMENT (i) | DISTANCE FROM CENTER LINE ($X_i$) |
|:---:|:---:|
| 1 | $0.0943R_B$ |
| 2 | $0.1371R_B$ |
| 3 | $0.1914R_B$ |
| 4 | $0.2657R_B$ |
| 5 | $0.3657R_B$ |
| 6 | $0.4914R_B$ |
| 7 | $0.6286R_B$ |
| 8 | $0.7714R_B$ |
| 9 | $0.9000R_B$ |
| 10 | $0.9829R_B$ |
| 11 | $0.9914R_B$ |
| 12 | $0.9429R_B$ |
| 13 | $0.8571R_B$ |
| 14 | $0.7543R_B$ |
| 15 | $0.6400R_B$ |
| $\Sigma$ | $9.0142R_B$ |
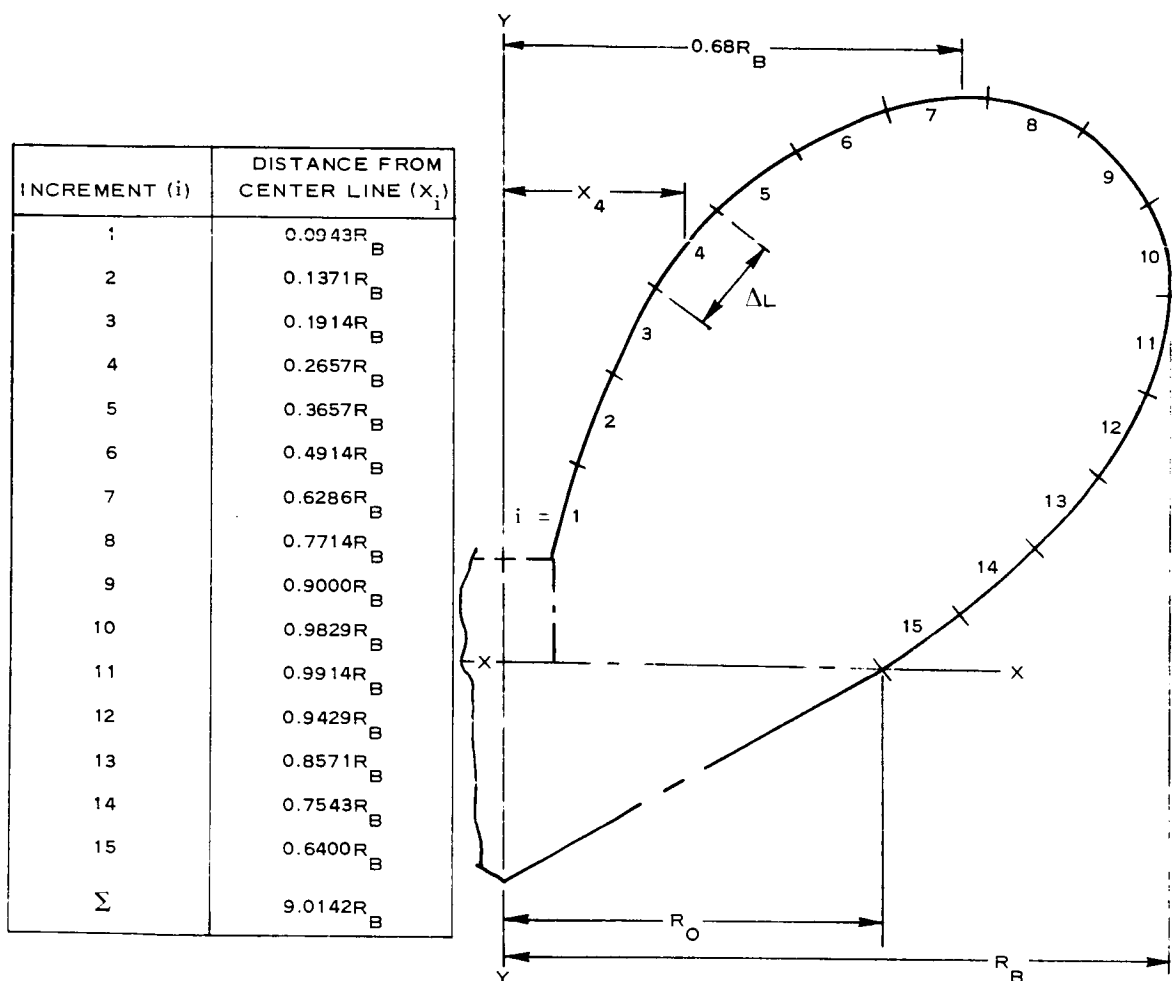
REAR HALF

$K = 0.95$

$\dfrac{\rho}{K} = 1.0$

$K = \rho = 0.475$

FRONT HALF

$K = 0.475$

$\Delta L = 0.143R_B$

$L_M = 15 (0.143R_B)$

$= 2.14R_B$

USE $L_M = 2.15 R_B$

Figure A-6 - Tucked-Back BALLUTE Profile

The surface area of a body of revolution is equal to the product of meridian length and centroidal perimeter. This can be expressed as

$$A_s = 2\pi(0.601R)(2.15R)$$

$$= 8.112R^2 . \tag{A-29}$$

The fabric stress is calculated at radius $R_B$ by noting that the rear half of the BALLUTE curve is tangent to a line parallel to the X axis at a value of $X = 0.68R_B$.

Summing forces in the Y direction produces

$$2f_f\pi R_B + nTm = \pi P\left[R_B^2 - (0.68R_B)^2\right]$$

or

$$2f_f + \frac{nTm}{\pi R_B} = R_B P(1 - 0.68^2) ,$$

when $f_f$ = fabric stress.

Thus, with

$$K = \frac{nTm}{P\pi R_B^2}$$

and

$$PKR_B = \frac{nTm}{\pi R_B} ,$$

fabric stress can be expressed as

$$f_f = \frac{PR_B}{2} (0.537 - K) . \tag{A-30}$$

Fabric weight is

$$W_f = \frac{f_f A_s}{K_f}$$

$$= \frac{8.12(R_B)^2 \left(\frac{PR_B}{2}\right)(0.537 - K)}{K_f}$$

$$= \frac{4.06 R_B{}^3 P(0.537 - K)}{K_f} \quad . \qquad \text{(A-31)}$$

Tension in a meridian is

$$T_m = \frac{K\pi R_B{}^2 P}{n}$$

and weight of the meridians is

$$W_c = \frac{n T_m L_m}{K_c}$$

$$= \frac{K\pi R^3 P 2.15}{K_c} \quad . \qquad \text{(A-32)}$$

Thus, BALLUTE weight is

$$W_B = W_f + W_c$$

$$= P\pi R_B{}^3 \left[\frac{1.29(0.537 - K)}{K_f} + \frac{2.15K}{K_c}\right]$$

$$= P\pi \frac{(d_B)^3}{8}\left[\frac{1.29(0.537 - K)}{K_f} + \frac{2.15K}{K_c}\right] . \qquad \text{(A-33)}$$

b.  AIRMAT Cone Decelerator

Weight equations for the AIRMAT cone decelerator are given below, based on geometry shown in Figure A-7.
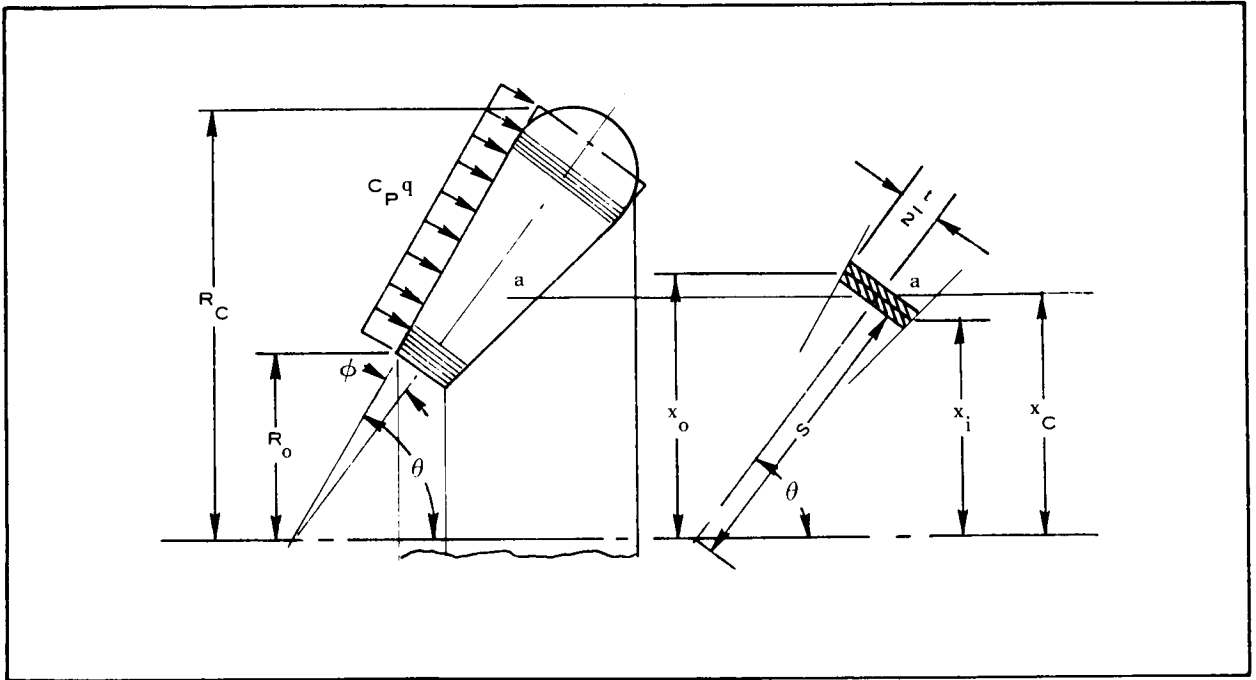
Figure A-7 - AIRMAT Cone Geometry

The weight of a pressurized isotensoid AIRMAT is given by the equation

$$W = \frac{3 \, pv \, FS}{K_m} \, .$$ 
(A-34)

Expressions v and p for the AIRMAT cone (see Figure A-7) are developed in the following manner:

$$\frac{t}{2} = S \tan \phi \, ,$$ 
(A-35)

$$S = \frac{x_c}{\sin (\theta - \phi)} \, ,$$ 
(A-36)

$$\frac{t}{2} = \frac{x_c \tan \phi}{\sin (\theta - \phi)} \, ,$$ 
(A-37)

$$x_o = x_c + \frac{t}{2} \cos(\theta - \phi)$$

$$= x_c \left[ 1 + \frac{\tan \phi}{\tan(\theta - \phi)} \right] , \qquad (A-38)$$

and

$$x_i = x_c - \frac{t}{2} \cos(\theta - \phi)$$

$$= x_c \left[ 1 - \frac{\tan \phi}{\tan(\theta - \phi)} \right] . \qquad (A-39)$$

The volume increment (dv) (see Figure A-8) can be expressed in the terms,

$$dv = 2\pi x_c t \, dS ,$$

$$dS = \frac{dx_c}{\sin(\theta - \phi)} ,$$

$$dv = \frac{4\pi x_c^2 \tan \phi \, dx_c}{\sin^2(\theta - \phi)} ,$$

and

$$v = \int_B^A \frac{4\pi x_c^2 \tan \phi \, dx_c}{\sin^2(\theta - \phi)} ,$$

when

$$A = \frac{R_c}{1 + \dfrac{\tan \phi}{\tan(\theta - \phi)}}$$

and

$$B = \frac{R_o}{1 + \dfrac{\tan \phi}{\tan(\theta - \phi)}} .$$
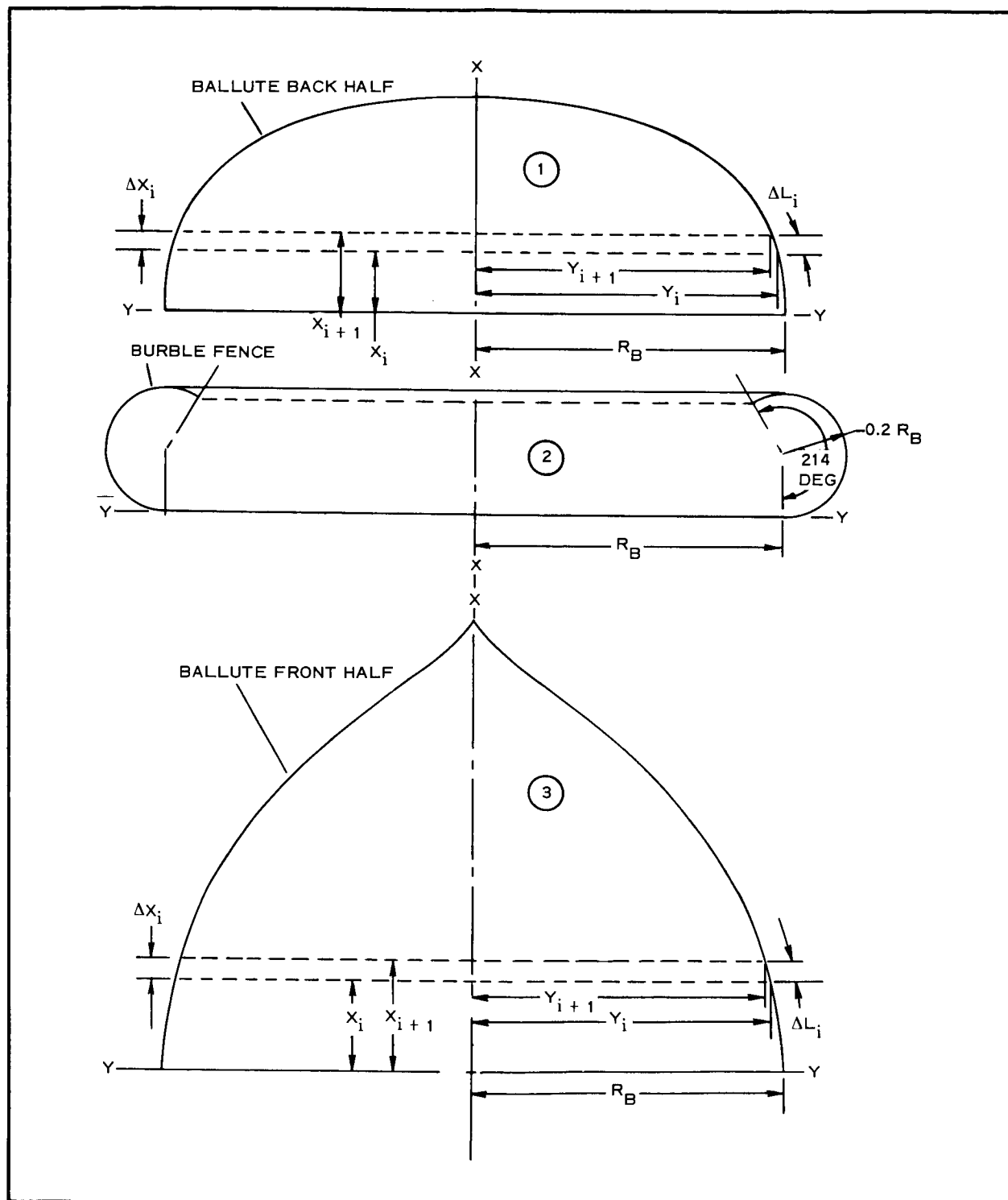
Figure A-8 - Typical BALLUTE

Thus,

$$v = \frac{4}{3} \frac{\pi \tan \phi}{\sin^2 (\theta - \phi)} \left[ \frac{\tan (\theta - \phi)}{\tan (\theta - \phi) + \tan \phi} \right]^3 (R_c^{\,3} - R_o^{\,3}) \; . \qquad \text{(A-40)}$$

Inflating to a pressure three times that required to support the external pressure and considering equilibrium of circumferential forces results in the following equations (see Figure A-7),

$$p_I t = \frac{3.0 C_p q x_o}{\cos \theta} \; ,$$

$$p_I = \frac{1.5 C_p q \left[ 1 + \dfrac{\tan \phi}{\tan (\theta - \phi)} \right] \sin (\theta - \phi)}{\tan \phi \cos \theta}$$

$$= 1.5 C_p q \frac{\left( \tan (\theta - \phi) + \tan \phi \right) \cos (\theta - \phi)}{\tan \phi \cos \phi} \; , \qquad \text{(A-41)}$$

and

$$p_I v = \frac{6 \pi C_p q}{3} \frac{\sin \theta \cos \phi - \cos \theta \sin \phi}{\cos \theta \left( \dfrac{\sin \theta}{\cos \theta} \right)^2} (R_c^{\,3} - R_o^{\,3})$$

$$= 2\pi C_p q \left( \frac{\cos \phi}{\sin \theta} \right)^2 \cos \phi \, (\tan \theta - \tan \phi)(R_c^{\,3} - R_o^{\,3}) \; , \qquad \text{(A-42)}$$

when

$$p_I = \text{internal pressure, and}$$

$$R_o = \text{reference radius.}$$

Substituting Equation A-42 into A-34 produces

$$W_C = \frac{6 \pi C_p q}{K_m} \left( \frac{\cos \phi}{\sin \theta} \right)^2 \cos \phi \, (\tan \theta - \tan \phi)(R_c^{\,3} - R_o^{\,3})(FS) \; .$$

To permit examination of cone weight in the same terms as BALLUTEs, let

$$P = C_p q \; ,$$

$$R = \frac{d}{2} \; ,$$

and

$$W_C = \frac{3\pi FSP}{4K_m} \left( \frac{\cos \phi}{\sin \theta} \right)^2 \cos \phi \; (\tan \theta - \tan \phi)(d_c^3 - d_o^3) \; . \qquad (A\text{-}43)$$

## 3. PHYSICAL CHARACTERISTICS ANALYSIS

### a. General

Physical characteristics of decelerators are defined as surface area, contained volume, centers of gravity of the surface and volume, and the inertias of the surface and volume. Typical configurations for the BALLUTE and tucked-back BALLUTE are used to calculate their physical characteristics. The shapes of typical bodies depend on some design parameters but generally will not differ enough from the typical profiles to affect analysis of characteristics significantly.

BALLUTE characteristics are functions of the ratio of a front cutoff radius to the maximum BALLUTE radius. For a radius ratio of zero, a complete BALLUTE usually is used as a trailing BALLUTE. BALLUTE inertias are about the corresponding centers of gravity.

A BALLUTE profile has three parts or curves, the rear half, the front half, and the burble fence (see Figure A-8). The Y-Y axis is common to all three, but the front half is rotated 180 deg about the Y-Y axis to join the three segments into a BALLUTE shape.

### b. BALLUTE Surface

The surface areas of the front and rear section are found by dividing profile arc lengths into equal lengths ($\Delta L$) (see Figure A-8). The radius to the centroid of the length increment is $\overline{Y}_i$. The surface areas are given by the equation: when $A'_{s\,1}$ = any BALLUTE part area.

$$A'_{s①} = 2\pi\Delta L \sum_1^n \bar{Y}_i ,$$  (A-44)

using the nondimensional forms: $\Delta L / R_B$ and $Y_i / R_B$,

$$A'_{s①} = 2\pi R_B^2 \frac{\Delta L}{R_B} \sum_1^n Y_i .$$

In the front half, n is the increment where the value of $Y_{(i + 1)} / R_B$ is $R_o / R_B$.

The burble-fence surface is

$$A'_{s②} = 2\pi L_② \bar{Y}_② .$$

For Figure A-8 proportions:

$$L_② = 0.748 R_B ,$$

$$\bar{Y}_② = 1.097 R_B ,$$

$$A'_{s②} = (1.097)(2\pi)(0.748) R_B^2$$

$$= 5.156 R_B^2 , \text{ and}$$

$$A_s = A'_{s①} + A'_{s②} + A'_{s③} .$$

The equation for the surface area is evaluated and may be written in the form

$$A_s = 10 C_S R_B^2 ,$$  (A-46)

where $C_S$ is a function of $R_o / R_B$ of the BALLUTE front half. $C_S$ is presented in Figure A-9. Surface centers of gravity of the rear and front halves of the BALLUTE are obtained from the equation for any of the three major elements of the BALLUTE,
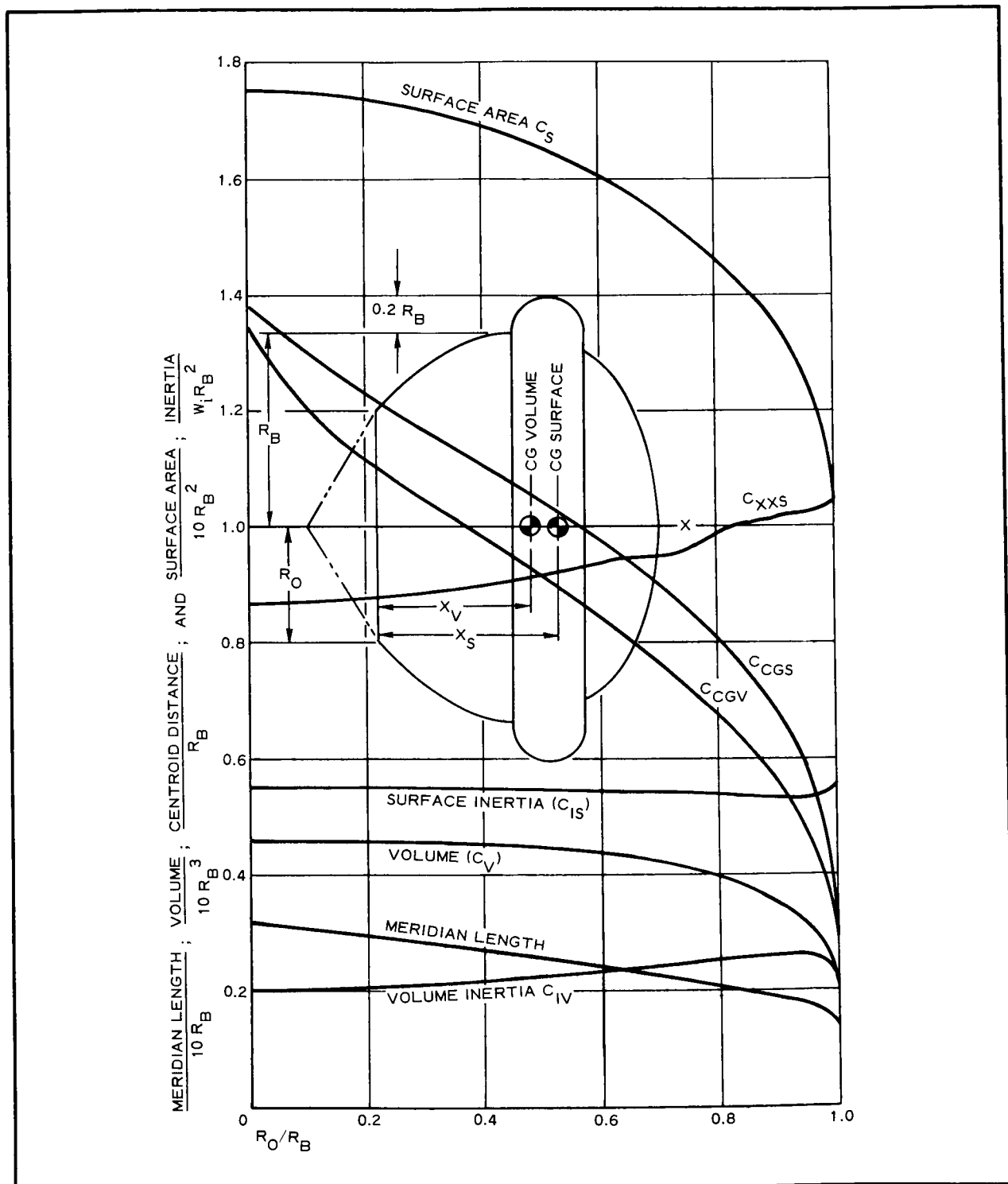
Figure A-9 - BALLUTE with 0.20 $R_B$ Burble Fence

$$\bar{X}_{①} = \frac{\sum_{1}^{n} \bar{Y}_i \bar{X}_i}{\sum_{1}^{n} \bar{Y}_i} . \tag{A-47}$$

Again for the front half n is the increment where the value of $Y_{i+1}/R_B$ is $R_o/R_B$.

The burble fence surface center of gravity of Figure A-8 is

$$X_{②} = 0.230 R_B . \tag{A-48}$$

The center of gravity of the complete BALLUTE is

$$X_s = \frac{A'_{s①} \bar{X}_{①} + A'_{s②} \bar{X}_{②} - A'_{s③} \bar{X}_{③}}{A_s} .$$

Then

$$\bar{X}_s = \bar{X}_{M③} - \bar{X}_s , \tag{A-49}$$

where $\bar{X}_{M③}$ is $\bar{X}_{i+1}$ of the front half corresponding to the value of n used in the equation for $A'_{s③}$.

The equation for $\bar{X}_s$ may be written in the form

$$\bar{X}_s = C_{CGS} R_B , \tag{A-50}$$

where $C_{CGS}$ is center of gravity of surface, a function of $R_o/R_B$ of the front half. $C_{CGS}$ is presented in Figure A-9.

The BALLUTE front and rear surface mass moments of inertia are calculated by treating each section $\Delta L$ as a thin hollow cylinder with radius equal to the average of $\bar{Y}_i$ and $\bar{Y}_{i+1}$, a length of $\Delta \bar{X}_i$ and being an average of $\bar{X}_i$ and $\bar{X}_{i+1}$ distance from the Y-Y axis. The surface mass moment of inertia about the Y-Y axis for any of the three elements is;

$$I_{YY\textcircled{1}} = \pi \gamma_s R_B^4 \frac{\Delta L}{R_B} \sum_1^n \left(\frac{\overline{Y}_{i\ avg}}{R_B}\right)^3 + \left(\frac{\overline{Y}_{i\ avg}}{6R_B}\right)\left(\frac{\Delta X_i}{R_B}\right) + 2\left(\frac{\overline{Y}_i}{R_B}\right)\left(\frac{\overline{X}_{i\ avg}}{R_B}\right)^2 ,$$

<div align="right">(A-51)</div>

where $\gamma$ is the average weight per square foot of the BALLUTE surface.

The burble fence of Figure A-8 surface mass moment of inertia about the Y-Y axis is:

$$I_{YY\textcircled{2}} = 3.854\gamma R_B^4 .$$

<div align="right">(A-52)</div>

The inertias of the three parts about the Y-Y axis are added and transferred to the surface center of gravity by standard inertia transfer equations. The surface mass moment of inertia may be written in the form

$$I_{CGS} = C_{IS} W_S R_B^2 ,$$

<div align="right">(A-53)</div>

where $W_S$ is the BALLUTE weight and $C_{IS}$ is a function of $R_o/R_B$ of the front half. $C_{IS}$ is also presented in Figure A-9.

The mass moment of inertia of the BALLUTE surface about the X-X axis is

$$I_{XXs} = \gamma_s R_B^2 \times$$

$$\left\{ \sum_1^{n_1} \Delta A'_{s\textcircled{1}}\left(\frac{Y_{avg\textcircled{1}}}{R_B}\right)^2 + \sum_1^{n_{\textcircled{3}}} \Delta A'_{s\textcircled{3}}\left(\frac{\overline{Y}_{avg\textcircled{3}}}{R_B}\right)^2 + A'_{s\textcircled{2}}\left(\frac{\overline{Y}_{\textcircled{2}}}{R_B}\right)^2 \right\}.$$

The value of $n_{\textcircled{3}}$ is the increment where the value of $R_o/R_B$ is identical to $\overline{Y}_{i+1}/R_B$. This equation may be written in the form

$$I_{XXS} = C_{XXS} R_B{}^2 W_S \ .$$

where $C_{XXS}$ is a function of $R_o/R_B$ of the front half of the BALLUTE. $C_{XXS}$ is presented in Figure A-9.

c. __BALLUTE Volume__

The volumes of the front and rear halves are found by dividing the profile x ordinates into a number of equal length increments, $\Delta X$.

The average radii of the lengths, $\bar{\bar{Y}}_{avg\ i}$, are calculated. The volumes are given by the equation:

$$V_i = \pi \Delta \bar{X}_i \sum_1^n (\bar{\bar{Y}}_{avg\ i})^2 \ . \qquad (A-54)$$

Using nondimensional forms, $\Delta \bar{X}/R_B$ and $\bar{Y}_{avg\ i}/R_B$,

$$V_i = \pi R_B{}^3 \frac{\Delta X}{R_B} \sum_1^n \left(\frac{\bar{\bar{Y}}_{avg}}{R_B}\right)^2 \ .$$

For the front half of the BALLUTE, n is the increment where the value of $\bar{Y}_{i+1}/R_B$ is $R_o/R_B$.

The burble fence volume is:

$$V_{②} = 2\pi \bar{Y}_V A \ . \qquad (A-55)$$

For Figure A-8 proportions,

$$\bar{Y}_V = 1.065 R_B \ ,$$

$$A = 0.07487 R_B{}^2 \ ,$$

$$V_{②} = 0.501 R_B{}^3 \ ,$$

and

$$\bar{X}_{V②} = 0.220 R_B \ . \qquad (A-56)$$

The volume of the BALLUTE in Figure A-8 is

$$V_B = V_① + V_② + V_③ .$$

The equation for the volume may be written in the form

$$V = 10 C_v R_B^3 , \qquad (A-57)$$

where $C_v$ is a function of $R_o/R_B$ of the BALLUTE front half. $C_v$ is presented in Figure A-9. Volume centers of gravity of the rear and front halves of the BALLUTE are obtained from the equation,

$$\bar{X}_{v①} = \frac{\displaystyle\sum_{1}^{n} \bar{Y}_{avg\ i}^2 \bar{X}_i}{\displaystyle\sum_{1}^{n} \bar{Y}_{avg\ i}^2} , \qquad (A-58)$$

where $\bar{X}_i$ is the $\bar{X}$ distance to the centroid of the $\Delta\bar{X}$ length increments. Again, for the front half n is an increment where the value of $Y_{i+1}/R_B$ is $R_o/R_B$. The burble fence volume center of gravity of Figure A-8 is given in Equation A-56.

The center of gravity of the complete BALLUTE volume is

$$X_v = \frac{V_① \bar{X}_{v①} + V_② \bar{X}_{v②} - V_③ \bar{X}_{v③}}{V} . \qquad (A-59)$$

Then

$$\bar{X}_v = \bar{X}_{M③} - \bar{X}_v ,$$

where $X_{M③}$ is the $X_{i+1}$ value of the front half which corresponds to the value of n used in the equation for $V_③$.

The equation for $X_v$ may be written in the form:

$$X_v = C_{CGv} R_B , \qquad (A-60)$$

where $C_{CGv}$ is a function of $R_o/R_B$ of the front half. $C_{CGv}$ is presented in Figure A-9.

The BALLUTE front and rear half volumes mass moment of inertia are calculated by considering each section $\Delta X$ as a cylinder with radius equal to the average of $\overline{Y}_i$ and $\overline{Y}_{i+1}$, a height of $\Delta \overline{X}$ and a distance of $\overline{X}_{vi}$ from the Y-Y axis. The volume mass moment of inertia about the Y-Y axis is:

$$I_{vYY\textcircled{1}} = \frac{\pi}{12} \gamma_G R_B^5 \frac{\Delta X}{R_B} \sum_1^n \left( \frac{Y_{avg\ i}}{R_B} \right)^2 \left\{ 3 \left( \frac{Y_{avg\ i}}{R_B} \right)^2 + \left( \frac{\Delta X}{R_B} \right)^2 + \left( \frac{\overline{X}_{vi}}{R_B} \right)^2 \right\},$$

$$(A-61)$$

where $\gamma_G$ is the gas weight.

For the burble fence of Figure A-8, the volume mass moment of inertia about the Y-Y axis is

$$I_{vYY\textcircled{2}} = 0.616 \gamma_G V_{\textcircled{2}} R_B^2. \qquad (A-62)$$

The inertias of the three parts of volume about the Y-Y axis are added and transferred to the volume center of gravity by standard inertia transfer equations. The volume mass moment of inertia may be written in the form:

$$I_{CGv} = C_{Iv} W_G R_B^2, \qquad (A-63)$$

where $W_G$ is the weight of the contained gas and $C_{Iv}$ is a function of $R_o/R_B$ of the front half. $C_{Iv}$ is presented in Figure A-9.

The mass moment of inertia of the contained gas is:

$$I_{XXv} = \gamma_G R_B^2 \left\{ \sum_1^n \frac{\Delta V_{\textcircled{1}}}{2} \left( \frac{Y_{avg\textcircled{1}}}{R_B} \right)^2 + \sum_1^n \frac{\Delta V_{\textcircled{3}}}{2} \left( \frac{Y_{avg\textcircled{3}}}{R_B} \right)^2 + V_{\textcircled{2}} \left( \frac{\overline{Y}_{v\textcircled{2}}}{R_B} \right)^2 \right\}.$$

$$(A-64)$$

This equation may be written in the reduced form

$$I_{XXv} = C_{XXv} W_G R_B^2 \; ,$$

where $C_{XXv}$ is a function of $R_o/R_B$ of the front half. $C_{XXv}$ is presented in Figure A-9. Physical characteristics of the tucked-back BALLUTE are obtained from the same equations used for the conventional BALLUTEs. These constants are:

$$C_S = 0.81 \; ,$$

$$C_{CGS} = 0.524 \; ,$$

$$C_{IS} = 0.384 \; ,$$

$$C_{XXS} = 0.630 \; ,$$

$$C_v = 0.1835 \; ,$$

$$C_{CGv} = 0.402 \; ,$$

$$C_{Iv} = 0.045 \; , \text{ and}$$

$$C_{XXv} = 0.443 \; .$$

Inertia properties of the AIRMAT cone shown in Figure A-7 are developed in the following manner, using Equation A-34 for weight and the derivation for Equation A-40 for volume.

The center of gravity of the AIRMAT cone is found by noting that

$$dW = \frac{3 P_I d_v FS}{K_m} \; ;$$

$$W_c Y = \int y_c \, dW \; ;$$

$$y_c = \frac{x_i}{\tan (\theta - \phi)} \; ;$$

$$W_c Y = \frac{3 P_I F S}{K_M} 4\pi \int_B^A \frac{x_c^2 \tan\theta \, dx_c}{\sin^2(\theta - \phi)} \, \frac{x_c}{\tan(\theta - \phi)}$$

$$= \frac{12 P_I F S \pi}{K_M} \, \frac{\tan\phi}{\tan(\theta - \phi) \sin^2(\theta - \phi)} \int_B^A x_c^3 dx_c$$

$$= \frac{12 P_I F S \pi}{K_M} \, \frac{\tan\phi}{\tan(\theta - \phi) \sin^2(\theta - \phi)} \left(\frac{x_c^4}{4}\right)\Bigg|_B^A \quad , \qquad (A-65)$$

when

$$A = \frac{R_c}{1 + \dfrac{\tan\phi}{\tan(\theta - \phi)}}$$

and

$$B = \frac{R_o}{1 + \dfrac{\tan\phi}{\tan(\theta - \phi)}} \quad ,$$

and

$$W \bar{Y} = \frac{3\pi P_I F S}{K_M} \, \frac{\tan\phi}{\tan(\theta - \phi) \sin^2(\theta - \phi)} \left[\frac{\tan(\theta - \phi)}{\tan(\theta - \phi) + \tan\phi}\right]^4 \left[R_c^4 - R_o^4\right].$$

Because

$$W = \frac{4\pi P_I \tan\phi \, FS}{K_M \sin^2(\theta - \phi)} \left[\frac{\tan(\theta - \phi)}{\tan(\theta - \phi) + \tan\phi}\right]^3 \left[R_c^3 - R_o^3\right] ,$$

$$\bar{Y} = \frac{3}{4 \tan(\theta - \phi) + \tan\phi} \left(\frac{R_c^4 - R_o^4}{R_c^3 - R_o^3}\right) . \qquad (A-66)$$

The mass moment of inertia about a diameter of a ring dS long having the volume dv and a weight dw is:

$$dI_{oRing} = \frac{dw R^2}{2} \ .$$
(A-67)

The inertia of the AIRMAT core about its center of gravity becomes

$$I_o = \int \frac{dw R^2}{2} + \int dw y^2 - W \bar{Y}^2$$

$$= \frac{12 P_I \pi F S}{K_M} \frac{\tan \phi}{\sin^2 (\theta - \phi)} \left\{ \int_B^A \left[ \frac{x_c^4}{4} + \frac{x_c^4}{\tan^2 (\theta - \phi)} \right] dx_c \right\} - W \bar{Y}^2 \ .$$
(A-68)

Thus,

$$\frac{I_o}{W} = \frac{3}{\left[ \frac{\tan (\theta - \phi)}{\tan (\theta - \phi) + \tan \phi} \right]^3 \left( R_c^3 - R_o^3 \right)} \left[ \frac{x_c^5}{10} + \frac{x_o^5}{5 \tan^2 (\theta - \phi)} \quad \Big|_B^A \right] -$$

$$\left\{ \left[ \frac{3}{4 \tan (\theta - \phi) + \tan \phi} \right] \left[ \frac{R_c^4 - R_o^4}{R_c^3 - R_o^3} \right] \right\}^2$$

$$= \frac{3}{\left| \tan (\theta - \phi) \tan \phi \right|^2} \left\{ \left[ \frac{R_c^5 - R_o^5}{R_c^3 - R_o^3} \right] \left[ \frac{\tan^2 (\theta - \phi)}{10} + \frac{1}{5} \right] - \left[ \frac{3}{16} \left( \frac{R_c^4 - R_o^4}{R_c^3 - R_o^3} \right)^2 \right] \right\} \ .$$
(A-69)

The polar moment of inertia of a ring dS long having a volume dv and a weight dW is

$$dI_{x\,ring} = dwR^2 \ .$$

(A-70)

Thus,

$$I_x = \frac{12\pi P_I FS}{K_M} \ \frac{\tan\phi}{\sin^2(\theta - \phi)} \ \int_B^A x_c^2 dx_c x_c^2 \ ,$$

and

$$\frac{I_x}{W} = \frac{3}{\left[\dfrac{\tan(\theta - \phi)}{\tan(\theta - \phi) + \tan\phi}\right]^2 \left[R_c^3 - R_o^3\right]} \ \left.\frac{x_c^5}{5}\right|_B^A$$

$$= \frac{3}{5}\left[\frac{\tan(\theta - \phi)}{\tan(\theta - \phi) + \tan\phi}\right]^2 \left(\frac{R_c^5 - R_o^5}{R_c^3 - R_o^3}\right) \ .$$

(A-71)

The weight may include the weight of the contained gas and the weight of the AIRMAT material.

## APPENDIX B - DYNAMIC ANALYSIS

This appendix describes a digital computer program for
simulating the motion of re-entry vehicles with either
rigidly erected or tethered decelerator systems. Eight
degrees-of-freedom equations of motion are developed
from basic principles. A Fortran IV program listing
and a computational flow chart are included.

## SECTION I

## INTRODUCTION

The primary objective of this computer program is to satisfy the
need for a trajectory program which meets the following requirement:

1.  Capable of simulating re-entry flight with both rigid and
    tethered decelerator systems.

2.  Capable of simulating the decelerator erection or inflation
    sequence.

3.  Capable of simulating entry into any planet; i.e. Earth,
    Mars, Venus, etc., with either spherical or oblate spheroidal
    shapes.

4.  Reasonable running time.

5.  Programmed in a computer language (Fortran IV) which is
    applicable to any scientific computer facility.

The differential equations of motion are based on an inertial rectangular
coordinate system with origin at the center of the earth.  The numerical
integrations of these equations are performed with the fourth order Runge
Kutta method using automatic step size control.

The aerodynamics include both body (re-entry vehicle) and tethered
decelerator effects separately.  The aerodynamic force and moment coeffic-
ients appear as linear table look-up functions of Mach number and angle
of attack.

# SECTION I

## INTRODUCTION
(Continued)

Matrix algebra is used almost exclusively.  The various matrix notations
are defined in Section III.

## SECTION II

### REFERENCE FRAMES AND TRANSFORMATIONS

The reference frames are right handed orthogonal sets.  They are designated by $X$ , $Y$ , $Z$ with corresponding unit vectors $\iota$ , $j$ , $k$ . The reference frames and their definitions are itemized below and shown in Figures I and II.

Inertial Reference Frame ( $X_I$ , $Y_I$ , $Z_I$ )

    1.  The origin $O_I$ coincides with the center of the earth[*]

    2.  $Z_I$ is parallel to the earth's rotational axis and is positive north.

    3.  $X_I$ and $Y_I$ lie in the earth's equatorial plane with the $X_I Z_I$ plane initially intersecting Greenwich, England.

Earth Reference Frame ( $X_E$ , $Y_E$ , $Z_E$ )

    1.  The origin $O_E$ coincides with the center of the earth.

    2.  $Z_E$ is parallel to the earth's rotational axis and is positive north.

    3.  $X_E$ and $Y_E$ lie in the earth's equatorial plane with $X_E$ passing through the Greenwich meridian.

Geocentric Reference Frame ( $X_G$ , $Y_G$ , $Z_G$ )

    1.  The origin $O_G$ coincides with the center of the earth.

- - - - - - - - - - - - - - - - - - - - - - - - - - - - - - - - - - - -

[*]Earth reference as used here implies any planet.

<div align="center">

SECTION II

REFERENCE FRAMES AND TRANSFORMATIONS
(Continued)

</div>

Geocentric Reference Frame (Continued)

  2. $X_G$ passes through the body centroid.

  3. $X_G Z_G$ plane contains the polar axis and $Z_G$ is positive
     north of the equatorial plane.

Body Geocentric Reference Frame ( $X_v$ , $Y_v$ , $Z_v$ )

  1. The origin $O_v$ is at the body centroid.

  2. The $Z_v$ axis passes through the earth's center.

  3. The $X_v Z_v$ plane contains the earth polar axis, $X_v$ is positive
     north and $Y_v$ is positive east.

Body Reference Frame ( $X_B$ , $Y_B$ , $Z_B$ )

  1. The origin $O_B$ is at the body centroid.

  2. The $X_B$ axis is colinear with the body centerline and is
     positive forward.

  3. $Y_B$ is positive out the right side and $Z_B$ is positive downward.

Tethered Decelerator Reference Frame ( $X_D$ , $Y_D$ , $Z_D$ )

  1. The origin $O_D$ is at the tether harness point.

  2. The $X_D$ axis lies along the tether line, positive toward body.

  3. $Y_D$ is positive out the right side and $Z_D$ is positive downward.

## SECTION II

### REFERENCE FRAMES AND TRANSFORMATIONS
#### (Continued)

A set of Euler angles $\psi$ , $\phi$ and $\theta$ establish the orientation of the body frame relative to the body geocentric reference frame. The sequential rotation starting from the inertial axis frame is:

1. The earth's sidereal rotation $\Omega_\epsilon t$ about the $Z_I$ axis to $X_E, Y_E, Z_E$.

$$\left[\Omega_\epsilon t\right] = \begin{bmatrix} \cos\Omega_\epsilon t & \sin\Omega_\epsilon t & 0 \\ -\sin\Omega_\epsilon t & \cos\Omega_\epsilon t & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

2. The longitudinal rotation $\psi$ about the $Z_E$ axis to $X_I, Y_I, Z_I$.

$$\left[\psi\right] = \begin{bmatrix} \cos\psi & \sin\psi & 0 \\ -\sin\psi & \cos\psi & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

3. The geocentric latitude rotation $-\phi$ about the $Y_I$ axis to $X_G, Y_G, Z_G$.

$$\left[-\phi\right] = \begin{bmatrix} \cos\phi & 0 & \sin\phi \\ 0 & 1 & 0 \\ -\sin\phi & 0 & \cos\phi \end{bmatrix}$$

SECTION II

REFERENCE FRAMES AND TRANSFORMATIONS
(Continued)

4.  A -90° rotation about the $Y_G$ axis to $X_V$, $Y_V$, $Z_V$.

$$[-90] = \begin{bmatrix} 0 & 0 & 1 \\ 0 & 1 & 0 \\ -1 & 0 & 0 \end{bmatrix}$$

5.  The Euler rotation $\psi$ about the $Z_Y$ axis to $X_2$, $Y_2$, $Z_2$.

$$[\psi] = \begin{bmatrix} \cos\psi & \sin\psi & 0 \\ -\sin\psi & \cos\psi & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

6.  The Euler rotation $\theta$ about the $Y_2$ axis to $X_3$, $Y_3$, $Z_3$.

$$[\theta] = \begin{bmatrix} \cos\theta & 0 & -\sin\theta \\ 0 & 1 & 0 \\ \sin\theta & 0 & \cos\theta \end{bmatrix}$$

7.  The Euler rotation $\phi$ about the $X_3$ axis to $X_B$, $Y_B$, $Z_B$.

$$[\phi] = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos\phi & \sin\phi \\ 0 & -\sin\phi & \cos\phi \end{bmatrix}$$

## SECTION II

### REFERENCE FRAMES AND TRANSFORMATIONS
(Continued)

8. The decelerator rotation $\zeta$ about the $Y_8$ axis to $X_4$, $Y_4$, $Z_4$.

$$\left[\zeta\right] = \begin{bmatrix} \cos\zeta & 0 & -\sin\zeta \\ 0 & 1 & 0 \\ \sin\zeta & 0 & \cos\zeta \end{bmatrix}$$

9. The decelerator rotation $-\eta$ about the $Z_4$ axis to $X_D$, $Y_D$, $Z_D$.

$$\left[-\eta\right] = \begin{bmatrix} \cos\eta & -\sin\eta & 0 \\ \sin\eta & \cos\eta & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

The matrix transformation between the reference frames are indicated by $\left[T\right]$. Subscripts are employed to indicate the two frames involved and the direction in which the transformation takes place. Thus, $\left[T_{E \to I}\right]$ indicates a transformation from the earth frame to the inertial frame. The inverse transformation is indicated by a -1 superscript and the transpose by a tilta ( $\sim$ ).

The transformations employed between the various reference frames in terms of single rotation matrices are,

## SECTION II

### REFERENCE FRAMES AND TRANSFORMATIONS
### (Continued)

1. Body geocentric to body reference:

$$\left[T_{V \to B}\right] = \left[\theta\right]\left[\phi\right]\left[\psi\right]$$

and

$$\left[T_{B \to V}\right] = \left[-\psi\right]\left[-\phi\right]\left[-\theta\right] = \left[\widetilde{T}_{V \to B}\right]$$

2. Geocentric to body reference:

$$\left[T_{G \to B}\right] = \left[T_{V \to B}\right]\left[-90\right]$$

and

$$\left[T_{B \to G}\right] = \left[90\right]\left[\widetilde{T}_{V \to B}\right] = \left[\widetilde{T}_{G \to B}\right]$$

where

$$\left[90\right] = \begin{bmatrix} 0 & 0 & -1 \\ 0 & 1 & 0 \\ 1 & 0 & 0 \end{bmatrix}$$

3. Body to decelerator reference:

$$\left[T_{B \to D}\right] = \left[-\eta\right]\left[\zeta\right]$$

and

$$\left[T_{D \to B}\right] = \left[-\zeta\right]\left[\eta\right] = \left[\widetilde{T}_{B \to D}\right]$$
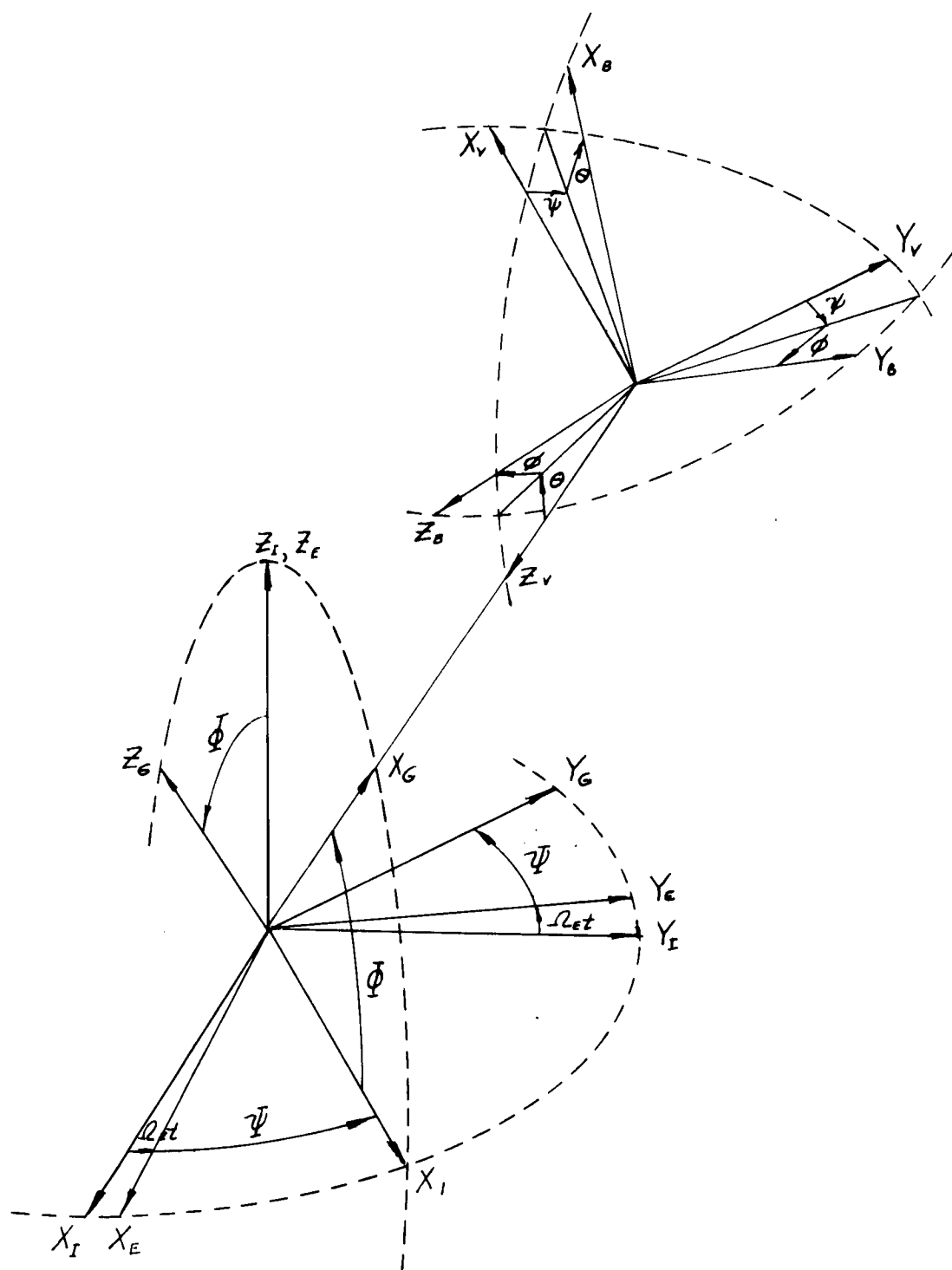
Reference Frames

Figure 1
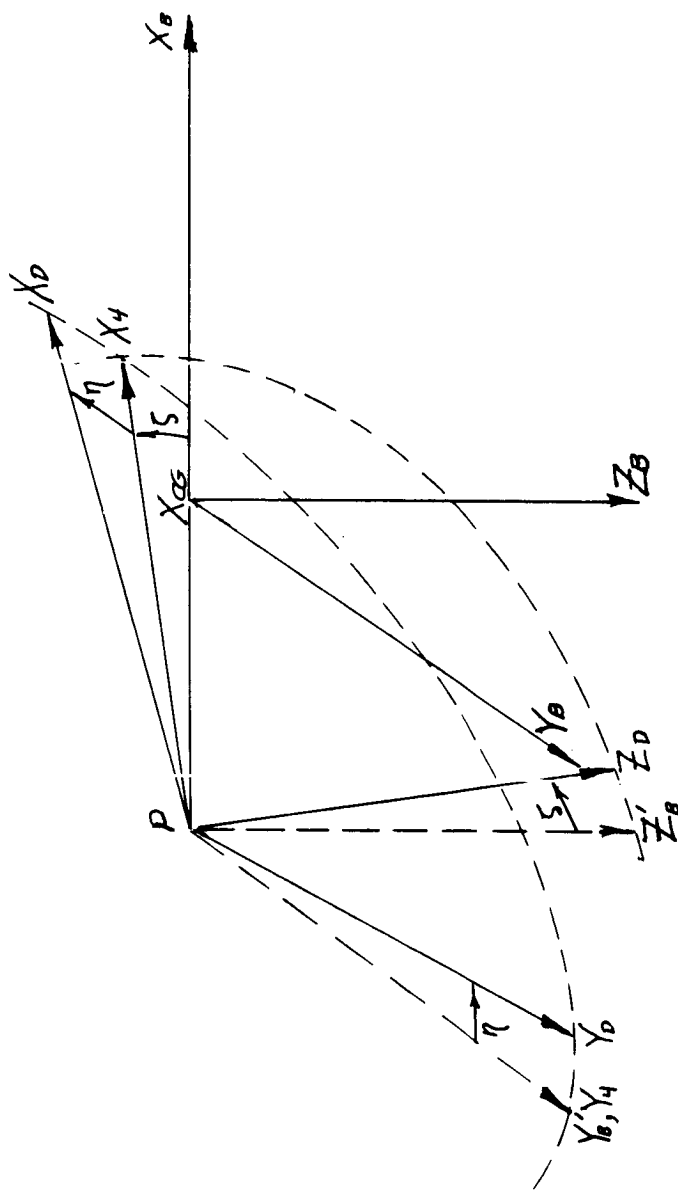
*FIGURE II*

*BODY AND TETHERED DECELERATOR REFERENCE FRAMES*

## SECTION III

## EQUATIONS OF MOTION

The equations of motion are derived from momentum considerations
(Newton's Second Law). The basic vector equations of motion for a mass
particle $\int m_p$ are:

$$\int \vec{F_I}(O)_B = \int m_p \frac{d \vec{V_I}(P)_B}{dt} \tag{1}$$

$$\int \vec{M_I}(P)_B = \frac{d}{dt} \int \vec{h_I}(P)_B \tag{2}$$

Where $\vec{V_I}(P)_B$ is the velocity vector and $\int \vec{h_I}(P)_B$ the momentum moment
vector of the mass particle.

The notation adopted is best explained by considering a general column
matrix $X_I(P)_B$ : $X$ indicates a coordinate matrix; $P$ indicates the
point the coordinate describes; the first subscript, in this case $I$ ,
indicates the coordinates are taken relative to the inertial frame and
the second subscript, in this case $B$ , indicates that the matrix components
are expressed in the body reference frame. The coordinate matrix $X_I(P)_B$
is written:

$$X_I(P)_B = \begin{bmatrix} X_I(P)_{i_B} \\ X_I(P)_{j_B} \\ X_I(P)_{k_B} \end{bmatrix}$$

SECTION III

EQUATIONS OF MOTION
(Continued)

The velocity and force matrices are similarly designated $V_I (P)_B$

and $F_I (P)_B$ which are the velocity and force of $P$ relative to the

inertial frame with components in the body frame.   $V_I (P)_B$ and $F_I (P)_B$

are written:

$$V_I (P)_B = \begin{bmatrix} V_I (P)_{i_B} \\ V_I (P)_{j_B} \\ V_I (P)_{k_B} \end{bmatrix} \quad ; \quad F_I (P)_B = \begin{bmatrix} F_I (P)_{i_B} \\ F_I (P)_{j_B} \\ F_I (P)_{k_B} \end{bmatrix}$$

An arrow is used to indicate a vector.  Thus, $\vec{V}_I (P)_B$ is the inertial

velocity vector of $P$ with components in the body system.

A.   Body Translational Equations

   Summing equation (1) over all the mass particles $\delta m$ of the body yields,

$$\sum \delta \vec{F}_I (P)_B \triangleq F_I (O)_B = \frac{d}{dt} \sum \vec{v}_I (P)_B \, \delta m \qquad (3)$$

where $F_I (O)_B$ is the external force matrix.  Assuming body

rigidity cancels the effects of internal forces since every particle

$(P)$ exhibits equal and opposite forces in accordance with Newton's

Third Law.

   The velocity of a mass particle $\delta m_P$ may be expressed as,

$$\vec{v}_I (P)_B = V_I (O)_B + \frac{d}{dt} \vec{X}_B (P)_B \qquad (4)$$

SECTION III

EQUATIONS OF MOTION
(Continued)

A.  Body Translational Equations (Continued)

where, $V_I (O)_B$ is the velocity vector matrix of the body centroid

and $\vec{X}_B (P)_B$ is the radius vector from the centroid to the particle.

Substituting this velocity relationship into the linear momentum

term of equation (3) and treating the body as a constant mass gives,

$$\sum \vec{v}_I (P)_B \, \delta m_P = \sum \left( V_I (O)_B + \frac{d}{dt} \vec{X}_B (P)_B \right) \delta m_P$$

$$= m \, V_I (O)_B + \frac{d}{dt} \vec{X}_B (P)_B \, \delta m_P$$

$$= m \, V_I (O)_B \qquad\qquad (5)$$

where $m$ is the total mass of the system of particles.  The summation

of $\vec{X}_B (P)_B \delta m_P$ vanishes since it represents the mass moment about the

centroid which by definition is zero.  Thus, equation (1) relates

the external forces to the motion of the center of mass.

$$F_I (O)_B = m \frac{d}{dt} V_I (O)_B = m \left\{ \dot{V}_I (O)_B + \left[ \omega_I (O)_B \times \right] V_I (O)_B \right\} \quad (6)$$

where $\dot{V}_I (O)_B$, as used here, is a scalar differentiation since

general vector integration is not definable.  The cross product

term $\left[ \omega_I (O)_B \times \right] V_I (O)_B$ constitutes the motion of the body frame

in inertial space.  This combined operation is vector differentiation

## SECTION III

## EQUATIONS OF MOTION
### (Continued)

A. **Body Translational Equations (Continued)**

and must be interpreted with care. Vector differentiation is further

explained in Appendix A of Reference 1.

The above equation can be expressed in a form suitable for integration

as follows:

$$\dot{V}_I (0)_B = \frac{1}{m} F_I (0)_B - [\omega_I (0)_B \times] V_I (0)_B \qquad (7)$$

The velocity relative to the inertial reference frame with components

in the body frame is found by integrating,

$$V_I (0)_B = \int_0^t \dot{V}_I (0)_B \, dt + V_I (0)_{B_0} \qquad (8)$$

B. **Body Rotational Equations**

The angular momentum of a particle ( $\int h_I (P)_B$ ) is the moment of its

linear momentum about a space fixed point. By locating this point

at the body centroid such that it is simultaneously a space fixed

and body fixed point, the summation of particle angular momentums

over the body results in,

$$\sum \int h_I (P)_B \triangleq H_I (0)_B = \sum ( \vec{X}_I (P)_B \times \vec{V}_I (P)_B ) \int_{m_p} \qquad (9)$$

## SECTION III

## EQUATIONS OF MOTION
### (Continued)


B. Body Rotational Equations (Continued)

Where $\vec{X}_I (P)_B$ is the radius vector from the centroid to the particle

mass $\delta m_p$ .


The particle velocity vector may be expanded to,


$$\vec{v}_I (P)_B = V_I (O)_B + [\omega_I (O)_B \times] \vec{X}_I (P)_B \qquad (10)$$


substituting for $\vec{v}_I (P)_B$ in equation (9).

$$H_I (O)_B = \sum \vec{X}_I (P)_B \times \left( V_I (O)_B + [\omega_I (O)_B \times] \vec{X}_I (P)_B \right) \delta m_p$$

$$= \sum \vec{X}_I (P)_B \times V_I (O)_B \, \delta m_p + \sum \vec{X}_I (P)_B \times [\omega_I (O)_B \times] \vec{X}_I (P)_B \, \delta m_p \qquad (11)$$

Since $V_I (O)_B$ is constant with respect to the summation and $\sum \vec{X}_I (P)_B \delta m_p \triangleq 0$

again by definition, the angular momentum matrix becomes,


$$H_I (O)_B = \sum \vec{X}_I (P)_B \times \omega_I (O)_B \times \vec{X}_I (P)_B \, \delta m_p$$

$$= \sum \left\{ \omega_I (O)_B \left( \vec{X}_I (P)_B \cdot \vec{X}_I (P)_B \right) - \vec{X}_I (P)_B \left( \omega_I (O)_B \cdot \vec{X}_I (P)_B \right) \right\} \delta m_p \qquad (12)$$

since $\vec{X}_I (P)_B$ has body components $X$ , $Y$ and $Z$ equation (12) may

SECTION III

EQUATIONS OF MOTION
(Continued)

B.  Body Rotational Equations (Continued)

also be written,

$$H_I(0)_B = \omega_I(0)_B \sum (X^2 + Y^2 + Z^2) \delta_{m_p} - \sum \chi_I(P)_B (PX + QY + RZ) \delta_{m_p}$$

where

$$\omega_I(0)_B \triangleq \begin{bmatrix} P \\ Q \\ R \end{bmatrix} \quad ; \quad \chi_I(P)_B \triangleq \begin{bmatrix} X \\ Y \\ Z \end{bmatrix}$$

The components of $H_I(0)_B$ are,

$$H_I(0)_{i_B} = P \sum (Y^2 + Z^2) \delta_{m_p} - Q \sum XY \delta_{m_p} - R \sum XZ \delta_{m_p}$$

$$H_I(0)_{j_B} = -P \sum XY \delta_{m_p} + Q \sum (X^2 + Z^2) \delta_{m_p} - R \sum YZ \delta_{m_p} \quad (13)$$

$$H_I(0)_{k_B} = -P \sum XZ \delta_{m_p} - Q \sum YZ \delta_{m_p} + R \sum (X^2 + Y^2) \delta_{m_p}$$

The summation $\sum (Y^2 + Z^2)$ is defined as the moment inertia $I_x$ of

the body mass about the $X$ axis. Similarly, the summation $\sum XY \delta_{m_p}$

is defined as the product of inertia $I_{xy}$ . The remaining summations

are similarly defined and equations (13) takes the form,

$$H_I(0)_B = \begin{bmatrix} I_x & -I_{xy} & -I_{xz} \\ -I_{yx} & I_y & -I_{yz} \\ -I_{zx} & -I_{zy} & I_z \end{bmatrix} \begin{bmatrix} P \\ Q \\ R \end{bmatrix} = [I] \, \omega_I(0)_B \qquad (14)$$

## SECTION III

### EQUATIONS OF MOTION
(Continued)

B.  **Body Rotational Equations (Continued)**

The angular momentum equation (equation 2) when summed over the

mass of the body may be written as,

$$M_I(0)_B = \frac{d}{dt} H_I(0)_B = \dot{H}_I(0)_B + [\omega_I(0)_B \times] H_I(0)_B \qquad (15)$$

where the dot indicates a scalar differentiation .  A more suitable

form for integration using the moment of inertia relationship of

expression (14) is,

$$M_I(0)_B = [I]\, \dot{\omega}_I(0)_B + [\omega_I(0)_B \times][I]\, \omega_I(0)_B \qquad (16)$$

In terms of the angular velocity in the body reference frame,

$$\dot{\omega}_I(0)_B = [I]^{-1} \left\{ M_I(0)_B - [\omega_I(0)_B \times][I]\, \omega_I(0)_B \right\} \qquad (17)$$

Integrating,

$$\omega_I(0)_B = \int_0^t \dot{\omega}_I(0)_B \, dt + \omega_I(0)_{B_o} \qquad (18)$$

The inverse moment of inertia matrix $[I]^{-1}$ presents a formidable

problem, not only above, but also in solving for the tension force in

the tether cable.  By simplifying the body shape to one which has

## SECTION III

## EQUATIONS OF MOTION
### (Continued)

B. **Body Rotational Equations (Continued)**

longitudinal symmetry the moment of inertia matrices are,

$$[I] = \begin{bmatrix} I_x & 0 & 0 \\ 0 & I_y & 0 \\ 0 & 0 & I_z \end{bmatrix} \quad ; \quad [I]^{-1} = \begin{bmatrix} 1/I_x & 0 & 0 \\ 0 & 1/I_y & 0 \\ 0 & 0 & 1/I_z \end{bmatrix} \quad (19)$$

C. **Euler Angle Equations**

The body angular orientation is described by the Euler angles $\psi$ , $\theta$ and $\phi$ (see Figure 1). These angles specify the ordered rotation between the body geocentric ( $V$ ) and the body ( $B$ ) reference frames and are obtained from angular rate relationships.

$$\vec{\omega}_I(0)_B = \vec{\dot{\psi}} + \vec{\dot{\theta}} + \vec{\dot{\phi}} + \vec{\dot{\phi}} + (\vec{\dot{\psi}} + \vec{\Omega_E})$$

In the matrix notation of Section I this relationship is,

$$\omega_I(0)_B = \begin{bmatrix} \dot{\phi} \\ 0 \\ 0 \end{bmatrix} + [\phi]\begin{bmatrix} 0 \\ \dot{\theta} \\ 0 \end{bmatrix} + [\phi][\theta]\begin{bmatrix} 0 \\ 0 \\ \dot{\psi} \end{bmatrix} + [T_{B \to V}]\begin{bmatrix} 0 \\ -\dot{\phi} \\ 0 \end{bmatrix} + [T_{B \to V}][-90][\phi]\begin{bmatrix} 0 \\ 0 \\ \dot{\psi} + \Omega_E \end{bmatrix}$$

$$= \begin{bmatrix} 1 & 0 & -\sin\theta \\ 0 & \cos\phi & \cos\theta\sin\phi \\ 0 & -\sin\phi & \cos\theta\cos\phi \end{bmatrix}\begin{bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} + [T_{V \to B}]\left\{ \begin{bmatrix} 0 \\ -\dot{\phi} \\ 0 \end{bmatrix} + [-90][\phi]\begin{bmatrix} 0 \\ 0 \\ \dot{\psi} + \Omega_E \end{bmatrix} \right\}$$

$$= [A][\dot{E}] + [T_{V \to B}]\begin{bmatrix} (\dot{\psi} + \Omega_E)\cos\phi \\ -\dot{\phi} \\ -(\dot{\psi} + \Omega_E)\sin\phi \end{bmatrix}$$

$$(20)$$

SECTION III

EQUATIONS OF MOTION
(Continued)

C. Euler Angle Equations (Continued)

The above relationship can be used in solving for the rates of

change of the Euler angles.

$$\left[\dot{E}\right] = \left[A\right]^{-1}\left[\omega_I(0)_B\right] - \left[A\right]^{-1}\left[T_{V \to B}\right]\begin{bmatrix} (\dot{\psi} + \Omega_E)\cos\bar{\phi} \\ -\dot{\phi} \\ -(\dot{\psi} + \Omega_E)\sin\bar{\phi} \end{bmatrix} \tag{21}$$

where,

$$\left[A\right]^{-1} = \begin{bmatrix} 1 & \dfrac{\sin\theta\sin\phi}{\cos\theta} & \dfrac{\sin\theta\cos\phi}{\cos\theta} \\ 0 & \cos\phi & -\sin\phi \\ 0 & \dfrac{\sin\phi}{\cos\theta} & \dfrac{\cos\phi}{\cos\theta} \end{bmatrix}$$

The Euler angles are obtained by integrating

$$\left[E\right] = \begin{bmatrix} \phi \\ \theta \\ \psi \end{bmatrix} = \int_0^t \left[\dot{E}\right] dt + \left[E\right]_0 \tag{22}$$

D. Tethered Decelerator Equations

The predominant motions of a tethered decelerator (Ballute type)

are represented by translational motion in spherical coordinates

with origin at the tether harness point ( $P$ ). Assuming direct

transmittal of body spin through the tether, the spin motions of

SECTION III

EQUATIONS OF MOTION
(Continued)

D. Tethered Decelerator Equations (Continued)

both body and decelerator may be combined in the body rotational

motion. Under this assumption, the tethered decelerator only adds

two degrees-of-freedom to the system. The added freedoms consist

of an angular pitch displacement ( $\zeta$ ) and a yaw displacement ( $\eta$ ),

(see Figure III).

The translational equation of motion (equation 6) yields an expression

for the decelerator motion.

$$
\begin{aligned}
F_I(d)_D &= m_D \frac{d}{dt} V_I(d)_D \\
&= m_D \left\{ \dot{V}_I(d)_D + \omega_I(0)_D \times V_I(d)_D \right\}
\end{aligned}
\tag{23}
$$

Where,

$$
\begin{aligned}
V_I(d)_D &= \left[ T_{B \to D} \right] \left\{ V_I(0)_B + \omega_I(0)_B \times X_B(P)_B \right\} \\
&\quad + \omega_I(0)_D \times X_D(d)_D \\
\dot{V}_I(d)_D &= \left[ \dot{T}_{B \to D} \right] \left\{ V_I(0)_B + \omega_I(0)_B \times X_B(P)_B \right\} \\
&\quad + \left[ T_{B \to D} \right] \left\{ \dot{V}_I(0)_B + \dot{\omega}_I(0)_B \times X_B(P)_B \right\} \\
&\quad + \dot{\omega}_I(0)_D \times X_D(d)_D \\
\omega_I(0)_D &= \left[ T_{B \to D} \right] \omega_I(0)_B + \omega_B(0)_D \\
\dot{\omega}_I(0)_D &= \left[ \dot{T}_{B \to D} \right] \omega_I(0)_B + \left[ T_{B \to D} \right] \dot{\omega}_I(0)_B + \dot{\omega}_B(0)_D
\end{aligned}
$$
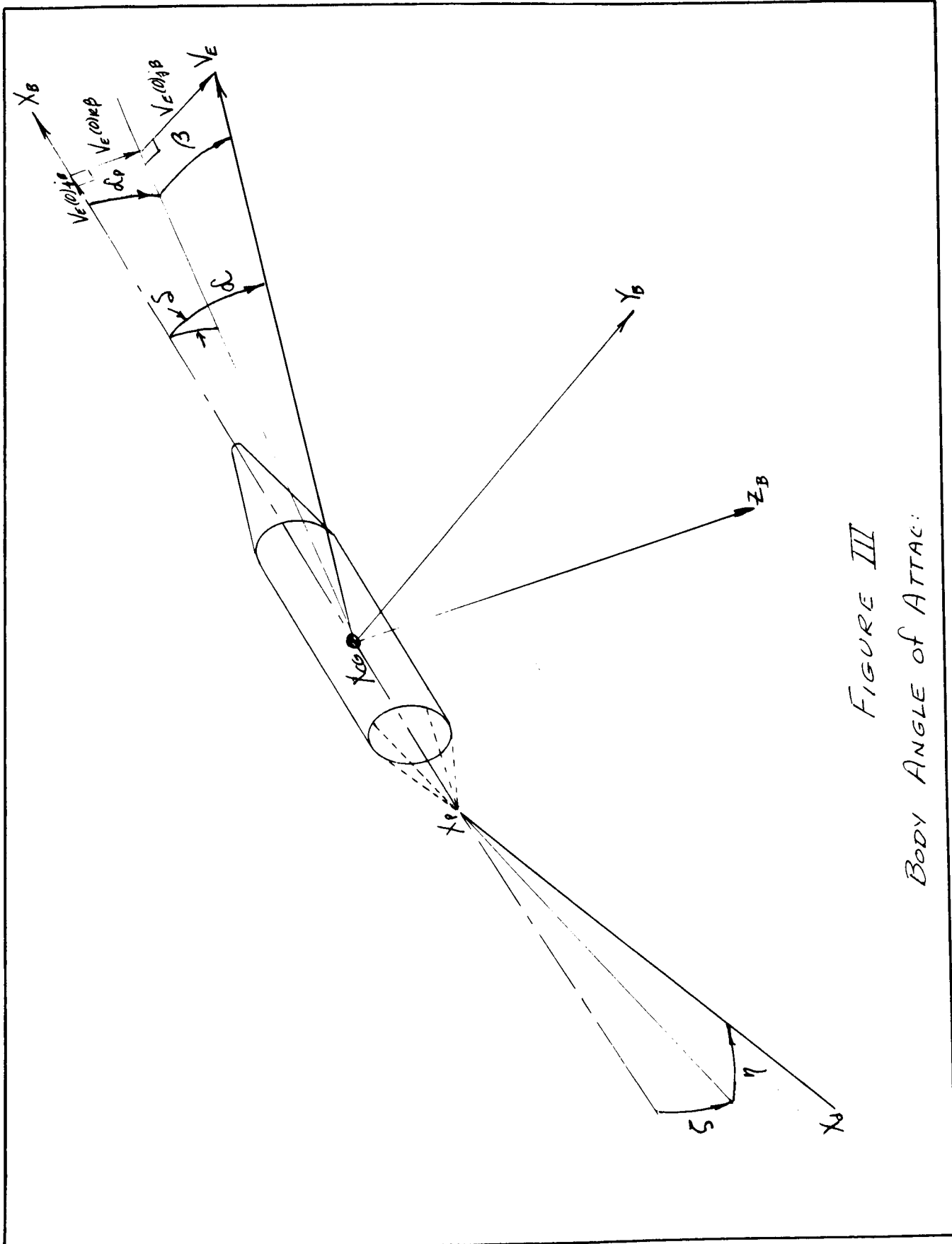
FIGURE III

BODY ANGLE OF ATTACK

# SECTION III

## EQUATIONS OF MOTION
### (Continued)

D. **Tethered Decelerator Equations (Continued)**

$$X_B (P)_B = \begin{bmatrix} -X_{\text{harness point}} \\ O \\ O \end{bmatrix} \quad ; \quad X_D (d)_D = \begin{bmatrix} -X_{\text{decelerator centroid}} \\ O \\ O \end{bmatrix}$$

$$[\dot{T}_{B \to D}] = \begin{bmatrix} -(\dot{\eta}\sin\eta\cos\zeta + \dot{\zeta}\cos\eta\sin\zeta) & -\dot{\eta}\cos\eta & \dot{\eta}\sin\eta\sin\zeta - \dot{\zeta}\cos\eta\cos\zeta \\ \dot{\eta}\cos\eta\cos\zeta - \dot{\zeta}\sin\eta\sin\zeta & -\dot{\eta}\sin\eta & -(\dot{\eta}\cos\eta\sin\zeta + \dot{\zeta}\sin\eta\cos\zeta) \\ \dot{\zeta}\cos\zeta & O & -\dot{\zeta}\sin\zeta \end{bmatrix}$$

substituting inertial and dynamic relationships for the acceleration

expressions $\dot{\omega}_I (O)_B$ and $\dot{V}_I (O)_B$ (See Section V) and expanding

$\dot{\omega}_B (O)_D \times X_D (d)_D$ in terms of $\ddot{\eta}$ and $\ddot{\zeta}$ the desired second order

differential parameters of the decelerator relative motion gives,

$$\frac{F_I^A (d)_D + F_I^T (O)_D + [T_{G \to D}] F_I^G (d)_G}{m_D} = [\dot{T}_{B \to D}] \left\{ V_I (O)_B + \omega_I (O)_B \times X_B (P)_B \right\}$$

$$+ [T_{B \to D}] \left\{ \frac{F_I^A (O)_B - [T_{D \to B}] F_I^T (O)_D + [T_{G \to B}] F_I^G (O)_G}{m} - \omega_I (O)_B \times V_I (O)_B \right.$$

$$+ [I]^{-1} \left[ M_I^A (O)_B + [T_{D \to B}] F_I^T (O)_D \times X_B (P)_B - \omega_I (O)_B \times [I] \omega_I (O)_B \right] \times X_B (P)_B \right\}$$

$$+ X_D \begin{bmatrix} O \\ \ddot{\eta} \\ \ddot{\zeta}\cos\eta - \dot{\zeta}\dot{\eta}\sin\eta \end{bmatrix} + [\dot{T}_{B \to D}] \omega_I (O)_B \times X_D (d)_D + [T_{B \to D}] \dot{\omega}_I (O)_B \times X_D (d)_D$$

$$+ \omega_I (O)_D \times V_I (d)_D$$

SECTION III

EQUATIONS OF MOTION
(Continued)

D.  Tethered Decelerator Equations (Continued)

Letting

$$[C] = -\left(\frac{F_I^A(d)_D + [T_{G \to D}] F_I^G(d)_G}{m_D}\right) + [\dot{T}_{B \to D}]\left\{V_I(o)_B + \omega_I(o)_B \times X_B(P)_B\right\}$$

$$+ [\dot{T}_{B \to D}] \omega_I(o)_B \times X_D(d)_D + [T_{B \to D}]\left\{\frac{F_I^A(o)_B + [T_{G \to B}] F_I^G(o)_G}{m} - \omega_I(o)_B \times V_I(o)_B\right.$$

$$\left. + [I]^{-1}\left[M_I^A(o)_B - \omega_I(o)_B \times [I] \omega_I(o)_B\right] \times X_B(P)_B\right\} + [T_{B \to D}]\left\{[I]^{-1}\left[M_I^A(o)_B\right.\right.$$

$$\left.\left. - \omega_I(o)_B \times [I] \omega_I(o)_B\right]\right\} \times X_D(d)_D + \omega_I(o)_D \times V_I(d)_D \qquad (25)$$

and,

$$[B] = \frac{F_I^T(o)_D}{m_D} + \frac{F_I^T(o)_D}{m} - [T_{B \to D}]\left\{[I]^{-1}\left([T_{D \to B}] F_I^T(o)_D \times X_B(P)_B\right) \times X_B(P)_B\right\}$$

$$- [T_{B \to D}]\left\{[I]^{-1}\left([T_{D \to B}] F_I^T(o)_D \times X_B(P)_B\right)\right\} \times X_D(d)_D$$

equation (24) may be written as,

$$[B] = [C] + X_D \begin{bmatrix} o \\ \ddot{\eta} \\ \ddot{\zeta}\cos\eta - \dot{\zeta}\dot{\eta}\sin\eta \end{bmatrix} \qquad (26)$$

The tension force magnitude may be isolated by employing the

following relationship:

$$F_I^T(o)_D = \begin{bmatrix} T \\ o \\ o \end{bmatrix} = T \begin{bmatrix} 1 \\ o \\ o \end{bmatrix} = T\left\{F^{T/T}(o)_D\right\}$$

# SECTION III

## EQUATIONS OF MOTION
### (Continued)

**D. Tethered Decelerator Equations (Continued)**

Thus, if

$$[B] = F_r^{T/r}(0)_D + F_r^{T/r}(0)_D - [T_{D \to D}]\left\{[I]^{-1}\left([T_{D \to B}]F_i^{T/r}(0)_D \times X_B(P)_B\right)\right. \times $$
$$\left. X_B(P)_B\right\}^{m_D} - [T_{B \to D}]\left\{[I]^{-1}\left([T_{D \to B}]F_r^{T/r}(0)_D \times X_B(P)_B\right)\right\}^{m} \times X_D(d)_D$$

then,

$$T[B] = [C] + X_D \begin{bmatrix} 0 \\ \ddot{\eta} \\ \ddot{\zeta}\cos\eta - \dot{\zeta}\dot{\eta}\sin\eta \end{bmatrix} \tag{27}$$

The tensile force magnitude (T) is given by,

$$T = \frac{[C]_i}{[B]_i} \tag{28}$$

The differential equations of relative decelerator motion are,

$$\ddot{\eta} = \frac{1}{X_D}\left\{T[B]_j - [C]_j\right\} \tag{29}$$

$$\ddot{\zeta} = \frac{\dot{\zeta}\dot{\eta}\sin\eta}{\cos\eta} + \frac{1}{X_D\cos\eta}\left\{T[B]_k - [C]_k\right\} \tag{30}$$

The following integrations define the motion.

$$\dot{\eta} = \int_0^t \ddot{\eta}\,dt + \dot{\eta}_0 \quad ; \quad \eta = \int_0^t \dot{\eta}\,dt + \eta_0 \tag{31}$$

$$\dot{\zeta} = \int_0^t \ddot{\zeta}\,dt + \dot{\zeta}_0 \quad ; \quad \zeta = \int_0^t \dot{\zeta}\,dt + \zeta_0 \tag{32}$$

## SECTION IV

## TRAJECTORY KINEMATICS

A.  Velocity and Positions

The motion of the body relative to the earth is measured in the

geocentric, $G$ , reference frame.  The relative earth velocity is

obtained from,

$$V_E (0)_G = \left[ T_{B \to G} \right] V_I (0)_B - \Omega_E (r)_G \tag{33}$$

where,

$$\Omega_E (r)_G = \begin{bmatrix} 0 \\ \Omega_E r \cos \phi \\ 0 \end{bmatrix} \quad ; \; r \text{ and } \phi \text{ are given by equation } (37).$$

The flight path is described by both an elevation angle ( $\gamma$ ) and an

azimuth angle ( $\lambda$ ).  The magnitudes of these angles are,

$$\gamma = \tan^{-1} \left\{ \frac{V_E (0)_{j_G}}{\sqrt{V_E (0)_{i_G}^2 + V_E (0)_{k_G}^2}} \right\} \tag{34}$$

and

$$\lambda = \tan^{-1} \left\{ \frac{V_E (0)_{j_G}}{V_E (0)_{i_G}} \right\} \tag{35}$$

The position of the body is described in the geometric terms of

longitude ( $\psi$ ), geocentric latitude ( $\phi$ ) and radial height ( $r$ )

by integrating the geocentric velocity expression $V_E (0)_G$ .

<center>

SECTION IV

TRAJECTORY KINEMATICS
(Continued)

</center>

A. Velocity and Positions (Continued)

$$[\dot{P}] = \begin{bmatrix} \dot{r} \\ \dot{\phi} \\ \dot{\psi} \end{bmatrix} = \begin{bmatrix} V_E(0)_{AG} \\ V_E(0)_{RG}/r \\ V_E(0)_{iG}\cos\phi/r \end{bmatrix} \tag{36}$$

$$[P] = \begin{bmatrix} r \\ \phi \\ \psi \end{bmatrix} = \int_0^t [\dot{P}]\,dt + [P]_0 \tag{37}$$

The position of the body and tethered decelerator are considered identical in the above coordinates. The velocities, however, may differ considerably. The relative earth velocity of the tethered decelerator is,

$$V_E(d)_G = \left[T_{B\to G}\right]\left\{ V_I(0)_B + \left[T_{D\to B}\right] V_B(d)_D \right\} \tag{38}$$

B. Angle of Attack

The angle of attack defines the relative orientation between the body axes (body reference frame) and the wind vector. Assuming that there are no disturbing winds and that the atmosphere rotates with the planet, the wind vector is coincident with the earth related velocity vector $V_E(0)_G$ .

The body angle of attack and its direction cosine are equated in

## SECTION IV

## TRAJECTORY KINEMATICS
### (Continued)

B.  Angle of Attack (Continued)

terms of the relative body velocity (see Figure III).

$$V_E(o)_B = \left[ T_{G \to B} \right] V_E(o)_G \tag{39}$$

$$\alpha = \tan^{-1} \left\{ \frac{\sqrt{V_E(o)_{B_j}^2 + V_E(o)_{B_k}^2}}{V_E(o)_{B_i}} \right\} \tag{40}$$

$$\delta = \tan^{-1} \left\{ \frac{V_E(o)_{B_j}}{V_E(o)_{B_k}} \right\}$$

$$\alpha_p = \tan^{-1} \left\{ \frac{V_E(o)_{B_k}}{V_E(o)_{B_i}} \right\} ; \quad \beta = \tan^{-1} \left\{ \frac{V_E(o)_{B_j}}{\sqrt{V_E(o)_{B_i}^2 + V_E(o)_{B_k}^2}} \right\}$$

The total tethered decelerator angle of attack relationships

(see Figure III) are defined in a similar manner.

$$V_E(d)_D = \left[ T_{G \to D} \right] V_E(d)_G \tag{41}$$

$$\alpha_D = \tan^{-1} \left\{ \frac{\sqrt{V_E(d)_{D_j}^2 + V_E(d)_{D_k}^2}}{V_E(d)_{D_i}} \right\} \tag{42}$$

$$\delta_D = \tan^{-1} \left\{ \frac{V_E(d)_{D_j}}{V_E(d)_{D_k}} \right\} \tag{43}$$

$$\alpha_{D_p} = \tan^{-1} \left\{ \frac{V_E(d)_{D_k}}{V_E(d)_{D_i}} \right\}$$

$$\beta_D = \tan^{-1} \left\{ \frac{V_E(d)_{D_j}}{\sqrt{V_E(d)_{D_i}^2 + V_E(d)_{D_k}^2}} \right\}$$

SECTION IV

TRAJECTORY KINEMATICS
(Continued)

C. Special Computations

The range traversed by the decelerator-body system over the planet surface is determined from the integral.

$$Range = \int_0^t R_\phi \sqrt{\dot{\phi}^2 + \dot{\psi}^2 \cos^2\phi} \, dt + Range_o$$  (44)

Where the oblate shape is approximated by:

$$R_\phi = R_o (1 - f \sin^2\phi)$$  (45)

$R_\phi$ — local planet radius

$R_o$ — equatorial radius

$f$ — oblateness parameter (earth = .0036700034)

For spherical planet considerations, the oblateness parameter is equal to zero and $R_\phi = R_o$.

The local altitude is determined from the relation,

$$h = r - R_o (1 - f \sin^2\phi)$$  (46)

The Mach number $M$ and dynamic pressure $q$ are given by

$$M = V_{E(O)_G} / V_s$$  (48)

$$q = \frac{1}{2} \rho \, V_{E(O)_G}^2$$  (49)

SECTION IV

TRAJECTORY KINEMATICS
(Continued)

C.  Special Computations (Continued)

where $V_s$ and $\rho$ are the atmospheric properties of the velocity of

sound and density respectively.  The Mach number and dynamic

pressure for the tethered decelerator is obtained by interchanging

$V_E(d)_c$ with $V_E(0)_G$ in the above expressions.  The atmospheric

properties are determined in a subprogram which may be changed to

incorporate any planetary atmosphere.  A sample Martian atmospheric

model is listed in Section VIII.

# SECTION V

## FORCES AND MOMENTS

The external forces are comprised of aerodynamic, gravity and cable

tension force components.  The force matrices are,

$$F_I(o)_B = F_I^A(o)_B - [T_{\cdot\,,\,}] F_I^T(o)_D + [T_{G\to B}] F_I^G(o)_G \tag{50}$$

$$F_I(d)_D = F_I^A(d)_D + F_I^T(o)_D + [T_{G\to D}] F_I^G(d)_G \tag{51}$$

where

$F_I^A(o)_B$ – body aerodynamic force matrix

$F_I^A(d)_D$ – decelerator aerodynamic force matrix

$F_I^G(o)_G$ – body gravitational force matrix

$F_I^G(d)_G$ – decelerator gravitational force matrix

$F_I^T(o)_D$ – cable tension force matrix

The cable tension force also contributes to the body moments.

$$M_I(o)_B = M_I^A(o)_B + M_I^T(o)_B \tag{52}$$

where
$$M_I^T(o)_B = [T_{D\to B}] F_I^T(o)_D \times X_B(P)_B$$

## A.  Aerodynamics

The aerodynamic body force matrix is

$$F_I^A(o)_{i_B} = C_A(M,\alpha)\, q\, S'$$

$$F_I^A(o)_{j_B} = -C_N(M,\alpha)\, \sin(S)\, q\, S' \tag{53}$$

$$F_I^A(o)_{k_B} = -C_N(M,\alpha)\, \cos(S)\, q\, S'$$

## SECTION V

### FORCES AND MOMENTS
### (Continued)

A. Aerodynamics (Continued)

and the body moment matrix is

$$M_I^A (0)_{B_i} = -C_{\ell_p}(M) \frac{W_I(0)_{B_i}}{|V_E(0)_G|} \frac{d^2}{2} q S \qquad (54)$$

$$M_I^A (0)_{B_j} = F_I^A(0)_{B_j} \left( \frac{X_{CP}(M,\alpha)}{d} - \frac{X_{CG}}{d} \right) d - C_{m_q}(M) \frac{W_I(0)_{B_j}}{|V_E(0)_G|} \frac{d^2}{2} q S$$

$$M_I^A(0)_{B_k} = -F_I^A(0)_{B_j} \left( \frac{X_{CP}(M,\alpha)}{d} - \frac{X_{CG}}{d} \right) d - C_{m_q}(M) \frac{W_I(0)_{B_k}}{|V_E(0)_G|} \frac{d^2}{2} q S$$

The decelerator aerodynamic forces are similarly defined:

$$F_I^A(d)_{D_i} = -C_{A_D}(M,\alpha) q_D S_D'(t) \qquad (55)$$

$$F_I^A(d)_{D_j} = -C_{N_D}(M,\alpha) \sin(\delta_D) q_D S_D'(t)$$

$$F_I^A(d)_{D_k} = -C_{N_D}(M,\alpha) \cos(\delta_D) q_D S_D'(t)$$

where

$S$ — aerodynamic reference area

$d$ — aerodynamic reference length

$C_A$ — aerodynamic axial force coefficient

$C_N$ — aerodynamic normal force coefficient

$C_{\ell_p}, C_{m_q}$ — aerodynamic moment damping coefficients

$X_{CP}$ — aerodynamic center of pressure along the longitudinal axis (distance from nose)

$X_{CG}$ — centroid location along the longitudinal axis (distance from nose)

## SECTION V

## FORCES AND MOMENTS
### (Continued)

B. Gravitation

The earth gravitational potential considering the second spherical

harmonic of the earth's oblateness is given by,

$$U(R,\phi) = \frac{GM}{R}\left\{ 1 + \frac{J_2 R_o^2}{2 R^2}(1 - 3\sin^2\phi) \right\} \tag{56}$$

where:

$G$  -  universal gravitational constant

$M$  -  mass of earth ($GM = 1.40786 \times 10^{16}$ ft$^3$/sec$^2$)

$R_o$  -  equatorial earth radius (20,926,759 ft)

$R$  -  geocentric radius of body

$\phi$  -  geocentric latitude

$J_2$  -  second harmonic term (1082.3 $\times 10^{-6}$)

The gravitational force matrix ($F_I^G(0)_G$) for the oblate earth is,

$$\frac{F_I^G(0)_G}{m} = \begin{bmatrix} -\frac{GM}{R^2}\left\{ 1 + \frac{3}{2}J_2\left(\frac{R_o}{R}\right)^2(1 - 3\sin^2\phi) \right\} \\ 0 \\ -\frac{GM}{R^2}\left( \frac{3}{2}J_2\left(\frac{R_o}{R}\right)^2 \right)^2 \cos\phi \sin\phi \end{bmatrix} \tag{57}$$

For spherical planet shapes

$$\frac{F_I^G(0)_G}{m} \begin{bmatrix} -g_o\left\{ \frac{R_o}{R} \right\}^2 \\ 0 \\ 0 \end{bmatrix} \tag{58}$$

where $g_o$ is the value of sea level acceleration due to gravity.

## SECTION VI

## DIGITAL TECHNIQUES AND EQUATION SOLUTIONS

A. **Matrix Routines**

Four matrix subroutines were developed for use in the program. These routines perform,

1. multiplication of a 3 x 3 matrix by another 3 x 3 matrix,

2. multiplication of a 3 x 3 matrix by a 3 x 1 column matrix,

3. multiplication using the transpose of a 3 x 3 matrix,

4. taking the cross product of two 3 x 1 column matrices.

The general matrix

$$X_B \, (P)_B = \begin{bmatrix} X \\ Y \\ Z \end{bmatrix}$$

is used to illustrate matrix operation notation. The cross product operator is,

$$\left[ X_B \, (P)_B \, X \right] = \begin{bmatrix} 0 & -Z & Y \\ Z & 0 & -X \\ -Y & X & 0 \end{bmatrix}$$

The transpose is,

$$\tilde{X}_B \, (P)_B = \begin{bmatrix} X & Y & Z \end{bmatrix}$$

The scalar derivative is,

$$\dot{X}_B \, (P)_B = \begin{bmatrix} \dot{X} \\ \dot{Y} \\ \dot{Z} \end{bmatrix}$$

SECTION VI

DIGITAL TECHNIQUES AND EQUATION SOLUTIONS
(Continued)

A. Matrix Routines (Continued)

The inverse and transpose relationships are,

$$[T_{V \to B}][T_{V \to B}]^{-1} = [1]$$ a unit matrix

$$[T_{V \to B}]^{-1} = [\widetilde{T}_{V \to B}] = [T_{B \to V}]$$ for orthogonal transformation

B. Numerical Integration

The fourth order Runge-Kutta numerical integration method is used
with Earnest's (Reference 3) step size control scheme. The sequence
of calculations performed in completing an integration step are

(1) $f_{io} = f_i(x_0, y_{10}, y_{20}, \cdots, y_{mo})$    $i = 1, 2, \cdots, m$

   $y_{i1} = y_{io} + \frac{h}{2} f_{io}$       $q_{i1} = f_{io}$

(2) $f_{i1} = f_i(x_0 + \frac{h}{2}, y_{11}, y_{21}, \cdots, y_{m1})$

   $y_{i2} = y_{io} + \frac{h}{2} f_{i1}$       $q_{i2} = q_{i1} + 2 f_{i1}$

(3) $f_{i2} = f_i(x_0 + \frac{h}{2}, y_{12}, y_{22}, \cdots, y_{m2})$                    (59)

   $y_{i3} = y_{io} + h f_{i2}$       $q_{i3} = q_{i2} + 2 f_{i2}$

(4) $f_{i3} = f_i(x_0 + h, y_{13}, y_{23}, \cdots, y_{m3})$

   $y_{i4} = y_{io} + \frac{h}{6}(q_{i3} + f_{i3})$   $q_{i4} = f_{i3}$

(5) $f_{i4} = f_i(x_0 + h, y_{i4}, y_{24}, \cdots, y_{m4})$

$$T_i = h(f_{i4} - q_{i3})$$                    (60)

SECTION VI

DIGITAL TECHNIQUES AND EQUATION SOLUTIONS
(Continued)

B.   Numerical Integration (Continued)

Where $\overline{T_i}$ is the error estimate. The error estimate corresponding

to the next larger step size $h^*$ is approximated by

$$\overline{T_i}^* = f \left( \frac{h^*}{h} \right)^4 \overline{T_i} \qquad\qquad (61)$$

Where $h$ is the current step size and $f$ is an input factor which

causes $\overline{T_i}^*$ to be overestimated to reduce time consuming premature

increases in step size.

An incremental step size specified by the program input determines

the amount of increase or decrease in ( $h$ ) the step size. If the

input factor is 2, the step size is halved if the truncation error

( $\overline{T_i}$ ) is too large and doubled if the error is less than a preset

magnitude limit ( $\overline{T_i}_{\ell_m}$ ) for a specified number of steps proves

helpful for rotational motions where $\overline{T_i}$ fluctuates.

Following is a flow chart showing the basic procedure.

## SECTION VI

## DIGITAL TECHNIQUES AND EQUATION SOLUTIONS
### (Continued)

B.  Numerical Integration (Continued)
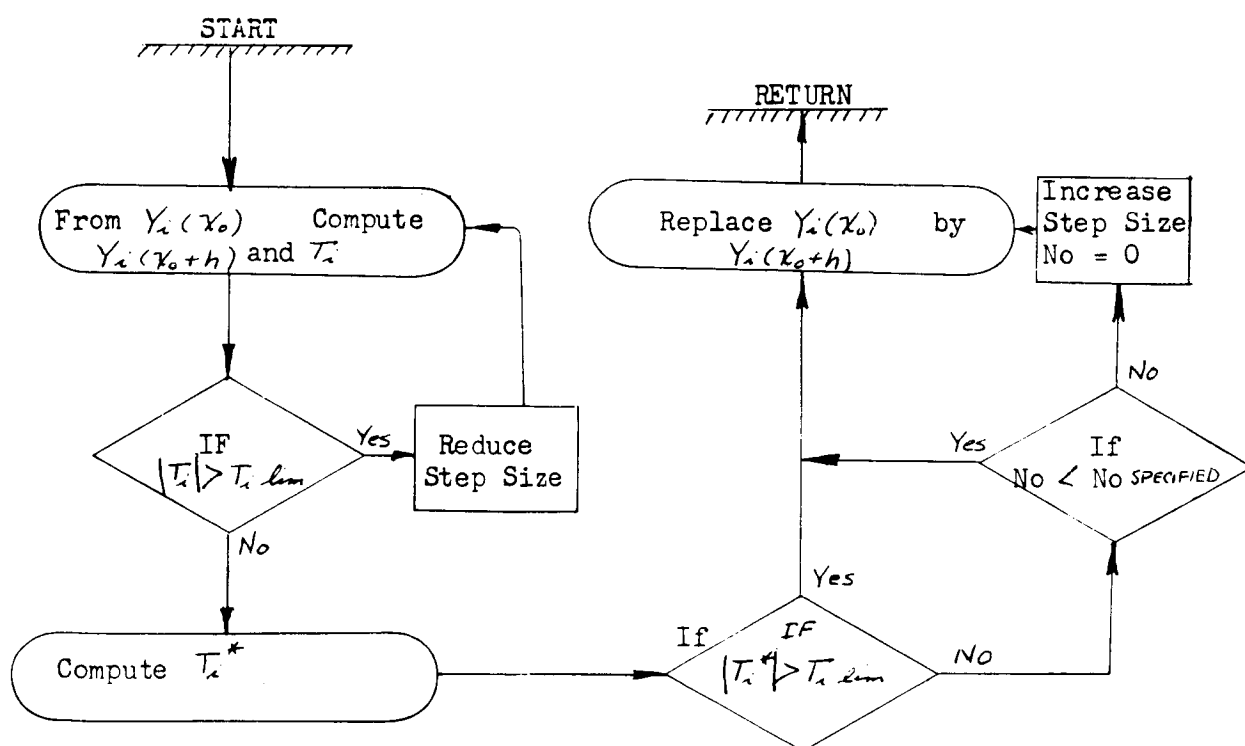


C.  Computational Flow Chart

On the following page is a simplified diagram of the computational

scheme for the complete computer program.

Start

Input

Initial Conditions

Output

Integration

Body
Reference
Transformations

Body
Velocity

Body
Angle
Of
Attack

Altitude
And
Range

Atmosphere

Decelerator
Gravitational
Force

Decelerator
Aerodynamics

Tether
Tension
Force

Decelerator
Angle of Attack

Body Force
And Moments
Including
Tether Force

Decelerator
Velocity

Decelerator
Equations
Of
Motion

Decelerator
Reference
Transformations

Body
Equations Of
Motion

No

Yes

If
Tethered
Decelerator

Translational

Rotational

Euler

Body
Forces
And
Moments

Body
Gravitational
Force

Body
Aerodynamics

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## SECTION VII

### COMPUTER INPUT-OUTPUT

</div>

An effort has been made to keep the input and output in conventional

parameters. The output consists of parametric histories.

A. Input Format

The following format gives the sequence of the input parameter, their

definitions and information concerning quantity, units, etc. The

Fortran IV input formats are; for real variables F10.3 and for integer

variables I5.

| Type | Symbol | Definition | Units |
|------|--------|-----------|-------|
| 1) | **Index Settings** | | |
| | I1 | Type of decelerator; 1-tethered, 0-rigid | |
| | I2 | No. of good integration steps before $\Delta t$ is increased | |
| | I3 | Planet shape; 1-oblate, 0-spherical | |
| | I4 | Symbolic unit for output | |
| | I5 | Symbolic unit for input | |
| | I6 | No. of values in Mach No. scale (1-8) | |
| | I7 | No. of values in angle of attack scale (1-8) | |
| | I8 | No. of Mach No. values used for decelerator | |
| | I9 | No. of angle of attack used for decelerator | |
| | I10 | No. of stacked trajectory runs | |
| | I11 | No. of differential equations 18-tethered 14-rigid | |
| | I12 | No. of negligible values to reduce from 18-14 DOF | |
| | I13 | No. of iterations per output | |
| | I14 | Inflation: 0-inflation, 1-inflated | |
| | I15 | Program parameter - must be zero | |
| | I16 | Program parameter - must be zero | |

SECTION VII

COMPUTER INPUT-OUTPUT
(Continued)

A. Input Format (Continued)

| Type | Symbol | Definition | Units |
|---|---|---|---|

2) Body Aerodynamics

$M$ — Mach No. table (16 values) — —

$\alpha$ — Angle of attack table (17 values) — deg.

$C_f$ — Body aerodynamic coefficient table
The following sequence is used:

 1. Read $C_A (M, \alpha)$ indexing Mach No. then angle of attack.

 2. Read $C_N (M, \alpha)$

 3. Read $X_{CP} (M, \alpha)$     from nose     ft.

$C_{\ell_p}$ — Roll damping coefficient $\left( C_{\ell_p} (M) \right)$ 18 values — sec/deg.

$C_{m_q}$ — Pitch and yaw damping coefficient $\left( C_{m_q} (M) \right)$    18 values — sec/deg.

3) Geophysics

$\mu$ — Universal gravitational term $(GM)$ — $ft^3/sec^2$

$R_o$ — Equatorial radius — ft.

$J_2$ — Second harmonic constant — —

$g_o$ — Sea level gravity — ft/sec.

$\Omega$ — Sidereal rotation rate of planet — rad/sec.

$f$ — Oblateness parameter — —

4) Position

$t$ — Time — sec.

$\psi$ — Geocentric longitude — deg.

$\phi$ — Geocentric latitude — deg.

$\varphi$ — Euler roll angle — deg.

$\theta$ — Euler pitch angle — deg.

$\psi$ — Euler yaw angle — deg.

$R_n$ — Range — n. mi.

## SECTION VII

## COMPUTER INPUT-OUTPUT
### (Continued)

### A. Input Format (Continued)

| Type | Symbol | Definition | Units |
|------|--------|-----------|-------|

5) **Body Rates and Orientation**

| | | | |
|---|---|---|---|
| $P$ | Body inertial roll rate | rad/sec. |
| $Q$ | Body inertial pitch rate | rad/sec. |
| $R$ | Body inertial yaw rate | rad/sec. |
| $\alpha$ | Angle of attack | deg. |
| $\beta$ | Angle of sideslip | deg. |
| $X_{CG}$ | Center of gravity location from nose | cal. |
| $Alt$ | Attitude | ft. |
| $Vel$ | Velocity magnitude | ft/sec. |

6) **Configuration**

| | | | |
|---|---|---|---|
| $m$ | Mass | slugs |
| $I_x$ | Moment of inertia | slug-ft$^2$ |
| $I_y$ | Moment of inertia | slug-ft$^2$ |
| $I_z$ | Moment of inertia | slug-ft$^2$ |
| $S$ | Body aerodynamic reference area | ft$^2$ |
| $d$ | Body aerodynamic reference length | ft. |
| $h_s$ | Altitude stop | ft. |
| $t_s$ | Time stop | sec. |

## SECTION VII

## COMPUTER INPUT-OUTPUT
(Continued)

### A. Input Format (Continued)

| Type | Symbol | Definition | Units |
|------|--------|------------|-------|
| 7) | Integration | | |
| | $\Delta t$ | Initial time increment | sec. |
| | $\Delta t_{min}$ | Minimum tolerable step size | sec. |
| | $\Delta t_{max}$ | Maximum tolerable step size | sec. |
| | $f$ | Integration step size factor (usually 2) | - |
| | $T_{\ell im}$ | Error estimate limit table sequance is: | |
| | | Range (1) | n. mi. |
| | | $[\epsilon]$ (3) | deg. |
| | | $[\eta, \varsigma]$ (2) | deg. |
| | | $[\dot{\eta}, \dot{\varsigma}]$ (2) | deg/sec. |
| | | $[P]$ (3) | ft. |
| | | $V_r(o)_B$ (3) | ft/sec. |
| | | $\omega_r(o)_B$ (3) | deg/sec. |

Tethered Decelerator (If Required; i.e., if I1 - 1)

| | Symbol | Definition | Units |
|---|--------|------------|-------|
| | $C_f$ | Decelerator aerodynamic coefficient table. The following input sequence is used: 1. Read $C_A(M, \alpha)$ indexing Mach No. then angle of attack. 2. Read $C_N(M, \alpha)$ | |
| | $X_P$ | Distance from body centroid to harness point | ft. |
| | $X_D$ | Tether cord length or distance from to decelerator centroid | ft. |
| | $m_D$ | Decelerator mass | slugs |
| | $\varsigma$ | Decelerator pitch angle | deg. |
| | $\eta$ | Decelerator yaw angle | deg. |
| | $\dot{\varsigma}$ | Decelerator pitch rate | deg/sec. |
| | $\dot{\eta}$ | Decelerator yaw rate | deg/sec. |
| | $\alpha_{DS}$ | Negligibility limit on decelerator pitching motion | - |
| | $t_t$ | Time table for deployment sequence (4 values) | sec. |
| | $S_t$ | Decelerator aerodynamic reference area in accordance with the above table. (4 values | ft$^2$ |

SECTION VII

COMPUTER INPUT-OUTPUT
(Continued)

A. Input Format (Continued)

The input data sheets are:

Payload

(1) Index:
Format (16I5)

I1 I2 I3 I4 I5 I6 I7 I8 I9 I10 I11 I12 I13 I14 I15 I16

— — — — — — — — — — — — — — — $O$ $O$

(2) Aerodynamics:
Format (8 F10.3)

$M_N$
$\alpha$
$C_A$

$C_N$

<u>SECTION VII</u>

<u>COMPUTER INPUT-OUTPUT</u>
<u>(Continued)</u>

A.   <u>Input Format (Continued)</u>

<u>Payload (Continued)</u>

(2)  Aerodynamics (Continued):

$\dfrac{CP}{D}$

     ___ ___ ___ ___ ___ ___ ___ ___
     ___ ___ ___ ___ ___ ___ ___ ___
     ___ ___ ___ ___ ___ ___ ___ ___
     ___ ___ ___ ___ ___ ___ ___ ___
     ___ ___ ___ ___ ___ ___ ___ ___
     ___ ___ ___ ___ ___ ___ ___ ___
     ___ ___ ___ ___ ___ ___ ___ ___
     ___ ___ ___ ___ ___ ___ ___ ___
     ___ ___ ___ ___ ___ ___ ___ ___
     ___ ___ ___ ___ ___ ___ ___ ___
     ___ ___ ___ ___ ___ ___ ___ ___

$C_{l_p}$   ___ ___ ___ ___ ___ ___ ___ ___

$C_{m_q}$   ___ ___ ___ ___ ___ ___ ___ ___

(3)  Geophysics:
     Format (8F10.3)

$\mu \qquad R_o \qquad J_2 \qquad g_o \qquad \Omega \qquad f$

___ ___ ___ ___ ___ ___

(4)  Position:
     Format (8F10.3)

$t \qquad \dot{\psi} \qquad \dot{\phi} \qquad \phi \qquad \theta \qquad \psi \qquad R_n$

___ ___ ___ ___ ___ ___ ___

(5)  Body Rates and Orientation:
     Format (8F10.3)

$P \qquad Q \qquad R \qquad \alpha \qquad \beta \qquad \chi_{cG} \qquad Alt \qquad Vel$

___ ___ ___ ___ ___ ___ ___ ___

## SECTION VII

### COMPUTER INPUT-OUTPUT
### (Continued)

A.    Input Format (Continued)

Payload (Continued)

(6)   Configuration:

Format (8F10.3)

$m$      $I_x$      $I_y$      $I_z$      $S$      $d$      $h_s$      $t_s$

___      ___      ___      ___      ___      ___      ___      ___

(7)   Integration:

Format (8F10.3)

$\Delta t$      $\Delta t_{min}$      $\Delta t_{max}$      $f$

___      ___      ___      ___

$u$      $V$      $W$      $P$      $Q$      $R$      $r$      $\bar{\phi}$

$T_{Lim}$

___      ___      ___      ___      ___      ___      ___      ___

$\bar{\psi}$      $\phi$      $\theta$      $\psi$      $R_n$      $\dot{\zeta}$      $\dot{\eta}$      $\zeta$

___      ___      ___      ___      ___      ___      ___      ___

$\eta$

___

## SECTION VII

### COMPUTER INPUT-OUTPUT
#### (Continued)

A. **Input Format (Continued)**

**Decelerator**

Format (8F10.3)

$C_A$

$C_N$

| $X_P$ | $X_D$ | $m_D$ | $\zeta$ | $\eta$ | $\dot{\zeta}$ | $\dot{\eta}$ | $\alpha_{DS}$ |
|-------|-------|-------|---------|--------|---------------|--------------|---------------|
| $t_{t_1}$ | $t_{t_2}$ | $t_{t_3}$ | $t_{t_4}$ | $S_{t_1}$ | $S_{t_2}$ | $S_{t_3}$ | $S_{t_4}$ |

SECTION VII

COMPUTER INPUT-OUTPUT
(Continued)

B.  Output Format

The program output consists of time histories of the various flight

parameters.  These flight parameters and their units are given below:

PAYLOAD

| TIME | ALT | VELOCITY | MACH NO. | GAMMA |
|------|------|----------|----------|-------|
| DT | LONG | U | VEL-S | ALPHA |
| MASS | LAT | V | DYN-PR | BETA |
| FN | RANGE | W | RHO | XCG |
| FX | MX | P | IX | PHI |
| FY | MY | Q | IY | THETA |
| FZ | MZ | R | IZ | PSI |

DECELERATOR

| TENSION | BETA | ALPHA | MASS | FA |
|---------|------|-------|------|-----|
| VELOCITY | ETA | ZETA | HARNESS-L | FN |
| MACH NO. | ETAD | ZETAD | DYN-PR | CORD-L |

PAYLOAD

| | | |
|------|---|---|
| TIME | - | Flight time (sec) |
| ALT | - | Altitude (ft) |
| VELOCITY | - | Relative earth velocity (ft/sec) |
| MACH NO. | - | Mach No. |
| GAMMA | - | Flight path elevation (deg) |
| DT | - | Time increment (sec) |
| LONG | - | Geocentric longitude (deg) |
| U | - | Relative velocity along body X axis (ft/sec) |
| VEL-S | - | Velocity of sound (ft/sec) |
| ALPHA | - | Angle of attack (deg) |
| MASS | - | Body mass (slugs) |
| LAT | - | Geocentric latitude (deg) |
| V | - | Relative velocity along body Y axis (ft/sec) |
| DYN-PR | - | Free stream dynamic pressure (lbs/ft$^2$) |
| BETA | - | Side slip angle (deg) |
| FN | - | Normal force (lbs) |

## SECTION VII

## COMPUTER INPUT-OUTPUT
### (Continued)

B. **Output Format (Continued)**

**PAYLOAD (Continued)**

| | | |
|---|---|---|
| RANGE | - | Range (n. mi.) |
| W | - | Relative velocity along body Z axis (ft/sec) |
| RHO | - | Density (slugs/ft$^3$) |
| XCG | - | Longitudinal centroid location (ft from nose) |
| FX | - | Body longitudinal aerodynamic force (lbs) |
| MX | - | Aerodynamic moment about body X axis (ft-lbs) |
| P | - | Angular rate about body X axis (deg/sec) |
| IX | - | Moment of inertia $I_x$ (slug-ft$^2$) |
| PHI | - | Euler bank angle (deg) |
| FY | - | Body normal aerodynamic force, pitch (lbs) |
| MY | - | Aerodynamic moment about body Y axis (ft-lbs) |
| Q | - | Angular rate about body Y axis (deg/sec) |
| IY | - | Moment of inertia $I_y$ (slug-ft$^2$) |
| THETA | - | Euler pitch angle (deg) |
| FZ | - | Body normal aerodynamic force, yaw (lbs) |
| MZ | - | Aerodynamic moment about body z axis (ft-lbs) |
| R | - | Angular rate about body Z axis (deg/sec) |
| IZ | - | Moment of Inertia $I_z$ (slug-ft$^2$) |
| PSI | - | Euler azimuth angle (deg) |

**DECELERATOR**

| | | |
|---|---|---|
| TENSION | - | Tether cord tension (lbs) |
| BETA | - | Angle of side slip (deg) |
| ALPHA | - | Angle of attack (deg) |
| MASS | - | Decelerator mass (slugs) |
| FA | - | Aerodynamic axial force (lbs) |
| VELOCITY | - | Relative (earth) velocity (ft/sec) |
| ETA | - | Pitch angle (deg) |
| ZETA | - | Yaw angle (deg) |
| HARNESS-L | - | Harness length, behind body CG (ft) |
| FN | - | Aerodynamic normal force (lbs) |
| MACH NO. | - | Mach Number |
| ETAD | - | Pitch angular rate (deg/sec) |
| ZETAD | - | Yaw angular rate (deg/sec) |
| DYN-PR | - | Decelerator dynamic pressure (lbs/ft$^2$) |
| CORD-L | - | Tether cord length to CG (ft) |

## NOMENCLATURE

| Symbol | Description |
|--------|-------------|
| $X, Y, Z$ | Position coordinates |
| $[E], \phi, \theta, \psi$ | Euler angles |
| $[P]; r, \Phi, \Psi$ | Geocentric planet coordinates; radial distance, latitude and longitude |
| $\Omega_E$ | Planet sidereal rotation rate |
| $\zeta, \eta$ | Tethered decelerator position angles |
| $\dot{\zeta}, \dot{\eta}$ | Tethered decelerator relative pitch and yaw rates |
| $H$ | Angular Momentum |
| $[T]$ | Matrix transformation |
| $F_I(o)_B$ | Sum of forces on the body; body axes components |
| $M_I(o)_B$ | Sum of moments on the body; body axes components |
| $m$ | Body mass |
| $I$ | Moment of inertia |
| $V_I(o)_B$ | Velocity of body relative to inertial space |
| $V_E(o)_B$ | Velocity of body relative to the earth |
| $\omega_I(o)_B$ | Inertial angular body velocity |
| $F_I(d)_D$ | Sum of forces on the tethered decelerator |
| $V_I(d)_D$ | Velocity of decelerator relative to inertial space |
| $\omega_I(o)_D$ | Inertial angular decelerator velocity |
| $V_E(d)_D$ | Velocity of decelerator relative to the earth |
| $T$ | Tether cable tension force |
| $\lambda$ | Flight path heading angle |
| $\gamma$ | Flight path elevation |

## NOMENCLATURE (Continued)

| Symbol | Description |
|---|---|
| $h$ | Altitude |
| $R_\phi$ | Local radius of earth |
| $\alpha$ | Angle of attack |
| $S$ | Angle of attack direction cosine |
| $C_A$ | Aerodynamic axial force coefficient |
| $C_N$ | Aerodynamic normal force coefficient |
| $X_{CP}$ | Aerodynamic longitudinal center of pressure location |
| $C_{\ell_p}, C_{m_q}$ | Aerodynamic damping moments |
| $g_o$ | Acceleration of gravity at sea level |
| $\rho$ | Atmospheric density |
| $V_s$ | Velocity of sound |
| $M$ | Mach Number |
| $q$ | Dynamic Pressure |
| $X_{CG}$ | Longitudinal location of centroid from nose |
| $X_P$ | Distance from body centroid to harness point |
| $X_d$ | Distance from harness point to decelerator centroid |
| $R_o$ | Planet equatorial radius |

# REFERENCES

1. Kolk, Richard W., "Modern Flight Dynamics," Prentice-Hall, Inc., Englewood Cliffs, N. J., 1961.

2. Birklein, K., "Preliminary Mars Trajectory Simulation Program," Goodyear Engineering Report No. GER 11525, March 1964.

3. Earnest, L. D., "Kutta Integration With Step Size Control," MIT Lincoln Laboratory.

4. Isakson, G., "Flight Simulation of Orbital Reentry Vehicles," ASD Technical Report 61-171(I), October 1961.

5. Schuler, J. M. and Pritchard, F. E., "Six Degrees-of-Freedom Equations of Motion for a Maneuvering Reentry Vehicle," Technical Report: NAVTRADEVCEN801A, U. S. Naval Training Device Center, Port Washington, New York, June 1962.

6. Dennison, A. J. and Butler, J. F., "Missile and Satellite Systems Program for IBM 7090," General Electric Report No. 61SD170, February 1962.

7. Schlemmer, J. W. and Campbell, D. E., "A General Six Degrees of Freedom Trajectory Model for an IBM 1410 Digital Computer," Goodyear Engineering Report No. GER 10614, March 1962.